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V/STOL HANDLING

II. DOCUMENTATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

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on

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II—Documentation

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ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

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AGARD Report No.577 Part II

V/STOL HANDLING-QUALITIES CRITERIA

II – Documentation

This report was prepared at the request of the Flight Mechanics Panel of AGARD – NATO.

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SUMMARY

This report prepared by a V/STOL Handling Qualities Committee sponsored by the AGARD Flight Mechanics Panel brings together a discussion of all the factors and the background data considered by the authors in preparing Part I, Criteria and Discussion of AGARD Report No. 577.

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TABLE OF CONTENTS

V/STOL HANDLING QUALITIES

II - DOCUMENTATION

INTRODUCTION

Background. The previous AGARD report on V/STOL handling qualities (ref. 1) and other earlier work on conventional aircraft handling qualities presented criteria and specifications with essentially no background or reference material. A small improvement in this respect was started in reference 2 but was very limited in scope. As a consequence, the user had little understanding of the limitations of the data on which the criteria or specifications were based nor could he evaluate the criteria with respect to their optimum application to a specific aircraft design. Further, without this background knowledge, there may be a tendency to write specifications for handling qualities which are too rigid, thereby lacking flexibility to provide tradeoffs or compromises that are always necessary in developing new aircraft.

This report (Part II) brings together a discussion of all the factors considered by the authors in preparing Part I, Criteria and Discussion of AGARD Report No. 577 (ref. 3). In this way, the information contained here will acquaint the reader with the philosophy and structure of the various criteria, resulting in a better appreciation for the rationale behind each particular criterion.

The criteria are based on several sources of information including analytical studies, piloted simulator tests, flight tests of VTOL and STOL aircraft (figs. 1 and 2), and from specially equipped variable stability aircraft and helicopters. A concerted effort was made to obtain data from several VTOL aircraft that had reached a more advanced state in Europe and each committee member was asked to supply data from tests conducted in his respective country. In this manner, the real value of AGARD participation would be brought out since with this cooperative approach more data could be obtained than if one country tried to gather information through normal channels. The STOL criteria were established primarily from research programs conducted in the U. S. on eight aircraft (see fig. 2 and ref. 4). These aircraft weighed from under 3,000 lb to over 150,000 lb with wing loadings from 23 to 46 lb/ft^2 and approach speeds from 40 to 90 knots.

For the most part, little quantitative handling qualities data were available from the VTOL aircraft tested in Europe, particularly for the more advanced types due to higher priorities placed on demonstration-type flights and limited funding. On the other hand, the majority of the STOL aircraft tested used propeller propulsion systems, and generalization of the results to jet-type STOL aircraft may be questionable in certain areas. In an overall view, it must be recognized that the bulk of background material originates from analytical, simulator, and test-bed-type aircraft, with little confirmation from operational-type aircraft. This is less serious for the purpose of establishing criteria, which are intended only to be typical; however, for specification purposes, greater effort is needed to obtain stability- and control-type data from production aircraft and operational experience must be obtained to firm up numbers in many areas.

In the following sections, an attempt has been made to introduce each criterion briefly along with a discussion of the background material for ease of reading, except for those areas where cross-referencing and the interrelationship of the criteria require a broader treatment.

TERMINOLOGY

Basic Terms.

VTOL: Vertical takeoff and landing. Pronounced by letter or as a word and used as an attributive adjective designating heavier-than-air aircraft.

STOL: Short takeoff and landing. Pronounced by letter or as a word and used as an attributive adjective designating heavier-than-air aircraft. The ground roll distance, or the total air distance to clear a given obstacle must be defined by the procuring agency.

V/STOL: Vertical and/or short takeoff and landing used as an attributive adjective to designate aircraft that have vertical and/or short takeoff capability.

VTO: Vertical takeoff. Pronounced by letter and used to describe a maneuver wherein a VTOL aircraft clears a given obstacle vertically.

STO: Short takeoff. Pronounced by letter and used to describe a maneuver wherein V/STOL aircraft clear a given obstacle in a minimum distance. Takeoff may be initiated from a hover, but forward speed is required to gain additional lift necessary to clear the obstacle or to operate outside the height-velocity envelope.

RTO: Rolling takeoff. Pronounced by letter and used to describe a maneuver wherein a ground roll distance is required to accelerate V/STOL aircraft to takeoff speed.

V_{app}: Approach speed. Used to define the stabilized landing approach speed that produces maximum performance and/or the landing approach speed for which the landing qualities criteria apply.

V_{con}: Conversion air speed. The air speed at which an accelerating transition is completed and the aircraft enters the aerodynamic flight regime.

V_{rec}: Reconverson air speed. The air speed at which a decelerating transition is initiated and the aircraft enters the power lift flight regime.

V_{min}: Minimum flight speed. The lowest steady flight speed at which all relevant handling criteria can be met with any given configuration and power setting. This includes configuration appropriate to takeoff,

landing approach (including both shallow and steep approach), landing and waveoff.

SAS: Stability augmentation system. Any system (e.g., mechanical, electrical, fluidic,) that supplements or modifies the stability and/or the damping of the basic aircraft.

IGE: In-ground effect.

OGE: Out-of-ground effect.

Flight Regimes.

Aerodynamic flight: Flight supported primarily by the forward flight dynamic pressure acting on non-revolving aerodynamic surfaces at air speeds above the power-off stall speed.

Power-lift flight: That flight regime of any aircraft where controlled, sustained flight is possible below the power-off stall speed and in which part or all of the lift and/or control moments are a function of power. (V/STOL handling-qualities criteria and specifications apply.)

Hover: To remain stationary relative to the air mass. Flight primarily supported by power plant(s) derived lift.

Spot hover: To remain stationary relative to a point on the ground. Flight primarily supported by power plants(s) derived lift.

Translation: Horizontal or vertical movement relative to a fixed point.

Accelerating transition: The act of flying an aircraft from the hover flight condition to the conversion air speed.

Decelerating transition: The act of flying an aircraft from the reconversion air speed to the hover flight condition.

Conversion: The act of making the configuration changes necessary to go from the appropriate takeoff configurations to the aerodynamic flight regime.

Reconversion: The act of making the configuration changes necessary to go from the aerodynamic flight regime to the appropriate landing configuration.

Transition envelope: That portion of the aircraft's flight envelope where steady, controllable flight is possible in the powered-lift flight regime. The envelope is defined by such factors as air speed, height, power, thrust vector angle, control margins, angle of attack, rate of climb or descent, etc.

Downwash: The downward component of the efflux from the lift system.

Ground effect: Any effect on the aircraft performance, stability, or control caused by the aircraft itself due to its proximity to the ground.

Recirculation: The phenomenon wherein a part of the lift system efflux is returned to flow through the lifting system again.

Reingestion: Recirculation of exhaust gases into the engine inlet(s), which usually occurs near the ground and can result in loss of power.

Height-velocity envelope: A family of curves that define the probability of aircraft damage on ground contact that would result if a failure of the power system occurred within a specific height velocity portion of the power-lift flight regime.

Reference speeds: Any speed (e.g., surface wind, landing approach speed, takeoff speed) that is specified by the procuring agency for which the handling-qualities criteria apply.

Control Systems.

Acceleration: A type of control system in which the steady-state acceleration is proportional to the control displacement. The control system variables pertinent to this system are control power and control sensitivity.

Rate damped: A type of control system that incorporates an angular rate feedback circuit to assist the pilot in controlling the aircraft. The control system variables pertinent to this system are control power, control sensitivity, and angular rate damping.

Rate command: A type of control system that causes the steady-state angular rate to be proportional to the appropriate control displacement by direct comparison of these two parameters. The pertinent system parameters are the same as for the rate-damped system.

Attitude command: A type of control system that causes the attitude changes to be proportional to the appropriate control displacement by direct comparison of these two parameters. The system variables pertinent to this system are control power, control sensitivity, angular rate damping, and attitude feedback gain.

Engineering Terms.

Breakout force: Total force required to initiate motion of the cockpit control, including friction, preload, and inertia of the control system.

Friction: A force that acts to resist control motion.

Preload: A control force, built into the control force feel system by pretensioning of a spring, that holds the trim control position selected by the pilot and/or assists the pilot in returning the cockpit control to trim position after a control input.

Free play: The lost motion between the cockpit control element and the movement of the moment-generating device.

Force gradient: That portion of the control force that opposes control displacement and returns the control to the trim position as the pilot input force is relaxed. It is the total force for a given control displacement, minus the breakout force including friction, divided by the control displacement, pounds per inch.

Control displacement: Control travel of the pilot's cockpit control element.

Control effectiveness: The ability of the control surfaces or moment generation devices to produce the moments commanded by the pilot that enables him to maintain and/or change the aircraft attitude, velocity and acceleration throughout the flight envelope.

Control sensitivity: The initial acceleration per unit step control displacement from a trim condition.

Control power: The maximum acceleration produced by a full step control displacement from a trim condition.

Control margin: The amount of control power remaining before, after, or during a maneuver that is available to the pilot to initiate a maneuver, recover from a maneuver, and/or correct for gust disturbances.

Rate damping: A moment-opposing motion, proportional to rate. For a first-order system, it is measured as the negative reciprocal of the time required to obtain 63 percent of the final steady-state angular velocity resulting from a step control input.

Damping ratio: Ratio of actual damping to critical damping. (Exponential attenuation envelope for oscillatory modes.)

Symbols.

g Gravitational constant, 32.2 fps/sec.

$T_{1/2}$ Time to damp to one-half amplitude, sec.

T_2 Time to double amplitude, sec.

Z_w Vertical velocity damping, 1/sec.

δ_a Aileron deflection, deg, or cockpit roll control input, in.

δ_e Elevator deflection, deg, or cockpit pitch control input, in.

δ_r Rudder deflection, deg, or cockpit yaw control input, in.

ζ Damping ratio

τ Time to 63 percent of steady-state value, sec.

σ Product of damping ratio and natural frequency ($\zeta\omega_n$), rad/sec.

ω_d Damped frequency, rad/sec.

ω_n Natural frequency, rad/sec.

Section 1.0

CHARACTERISTICS OF THE CONTROL SYSTEMS

1.1 GENERAL

Experience has shown that the mechanical characteristics of V/STOL control systems are more critical than those for conventional aircraft. The wide variety of V/STOL concepts (tilt wing, jet lift, fan-in-wing, etc.) have used different types of control systems. In almost all these systems, a given cockpit control must move more than one moment-producing device, that is, the lateral control may govern propeller pitch and aileron and the longitudinal control may produce changes in the elevator position plus tail prop angle. Second, the V/STOL control system must have satisfactory sensitivity for hover as well as high-speed flight. The criteria of Part I and the documentation in this report consider several basic types of control systems (i.e., attitude, rate damped, and acceleration). In so doing, an attempt was made to establish criteria that would produce satisfactory handling qualities and still permit compromises for control travel, control effective range, control forces, etc. Control system characteristics of several V/STOL aircraft are summarized in figures 3-5.

1.2 CONTROL BREAKOUT FORCES

The control breakout force criteria of section 1.2 (ref. 3) are presented in table 1.1.

Table 1.1

V/STOL Control Breakout Force Criteria

Control axis	Acceleration system, lb	Rate system, lb	Attitude system, lb	After failure of power control system, lb
Pitch	0.5-1.5	0.5-3	0.5-3	5
Roll	0.5-1.5	0.5-3	0.5-3	4
Yaw	1-10	1-10	-	15
Height				
collective	1-3	1-3	-	5
throttle	1-3	1-3	-	3

The values shown in table 1.1 were obtained from an unpublished control force study, conducted on a piloted six-degree-of-freedom motion simulator, and from flight-test reports of V/STOL aircraft. These data and the breakout force criteria limits are graphically presented in figures 6, 7, and 8.

Some breakout force is required for all types of controlling methods to prevent inadvertent inputs when the pilot's hand is replaced on the controls. However, the force must not be too great or it will impede movement of the control, especially important with an acceleration system where small, precise inputs are required. The maximum roll and pitch breakout forces are set at 3 lb for the rate and attitude systems, but only 1.5 lb for the acceleration system. Insufficient data exist to completely justify this assumption.

Reference 5 shows that the directional breakout force for the XC-142 aircraft varied randomly from 0 to 10 lb at different wing angles. Pilots rated these characteristics undesirable and reported a preference for a constant breakout force for all phases of flight without specifying an optimum value.

Most height control systems have adjustable friction that permits the pilot to vary breakout force to match the flight condition, pilot seating position, and pilot strength. Existing data, therefore, do not contain documentation for breakout forces that are too high since high forces were immediately readjusted to desirable values.

Reference 6 reports that the CL-84 aircraft had a throttle breakout force of 1 lb which was judged to be satisfactory. Reference 5 reports that the pilots preferred a collective breakout force setting of approximately 3 lb. Unless a collective lever is mass-balanced at its pivot point, it can be affected by normal acceleration and a different breakout force setting is required for high transition speeds compared to hover. It is not clear in reference 5 if the 3-lb force was satisfactory for all powered-lift flight regimes or just for the hover. Reference 7 reports that the throttle quadrant of the P1127 aircraft was equipped with an adjustable friction lock or damper whose proper adjustment prior to VTOL flight was a matter of some importance. In this aircraft, it was necessary to release the throttle (height control) to change vector angle. If the friction setting was set too low, the throttle would creep aft, thus reducing power. The pilot's recommendation (ref. 7) was for a minimum level of friction that would prevent throttle creep, but the quantitative level was not specified. Another solution would be a friction device that permits the pilot to adjust the force without removing his hand from the throttle, which was not possible in the P1127. (Additional discussion on this subject is contained in section 1.8, Characteristics of Thrust Vector Controls.)

1.3 CONTROL FORCE GRADIENTS

The control force gradient criteria of section 1.3 (ref. 3) are presented in table 1.2.

Table 1.2			
Control Force Gradients for Hover			
Control axis	Type of Control System		
	Acceleration system, lb/in.	Rate system, lb/in.	Attitude system, lb/in.
Pitch	1-2.5	1-3.5	1-3
Roll	0.5-1.25	0.5-1.75	0.5-1.5
Yaw	2.5-10	2.5-10	-
Note: After a failure in a powered-control system, the gradients should not be more than twice the above values.			

The hover control force gradients (table 1.2) were obtained from a control force study utilizing a piloted six-degree-of-freedom motion simulator. This study was limited to the lateral axis and was conducted to determine the effect of various force gradients on handling qualities in hover. Results of this study are presented in figures 9-12 for an acceleration, rate system and two attitude systems. This was a very limited study using only two pilots and did not fully explore large ranges of each variable. It was, however, the only systematic control force study available that studied acceleration, rate, and attitude control systems. As shown in figures 9-12, pilot rating curves were drawn for each pilot and each system. Because of the limited nature of this study, it was not possible to draw 3-1/2 and 6-1/2 pilot rating boundaries. The optimum boundaries simply show the range of lateral force gradient that satisfied both pilots. In other words, if a lateral control system had a gradient within the optimum boundaries, it would not be objectionable to either pilot.

The longitudinal and directional force gradient criteria were established by multiplying the lateral optimum boundaries by a control force ratio. (This ratio will be discussed in more detail in a following paragraph.) After establishing longitudinal, lateral, and directional force gradient limits (fig. 6-8), force characteristics from various VTOL aircraft were plotted to show the agreement between criteria limits, actual forces, and pilot opinions. Pilot ratings of these actual force characteristics are presented in table 1.3.

Table 1.3			
Pilot Rating of Hover Force Characteristics			
Aircraft	Numerical rating	Pilot comments	Reference figure
XV-5A	3	Longitudinal and lateral forces too high for low-speed maneuvers. Low directional breakout and gradient forces produce a no "feel" system in hover.	6, 7
P1127	Satisfactory	Heavier longitudinal forces would have degraded handling qualities.	6
CL-84	Satisfactory	Hover force gradients were satisfactory. Difficulty encountered in stabilizing pitch attitude at intermediate wing angle positions may have been augmented by light force gradients.	6
XC-142	Dir. 4	Directional force gradients were too high. Should be reduced by at least 50%. Poor harmony with other axes.	8
SG-1262 Hover rig		Satisfactory	6-8

The roll control force gradients of the SG-1262 hover rig (ref. 8) exceeded the limits of the criteria shown in table 1.2 after the first inch of travel (fig. 7) but were still rated satisfactory. These data for the SG-1262 hover rig are somewhat misleading since the 3 lb/inch roll control force gradient is twice the maximum specified. This is possible because control gearing was nonlinear so that maximum roll control power was obtained within the first inch of control travel (see section 1.9). Pilots, unaware of this feature, complained about the roll control force required to maintain the limited bank angle (±15°) because they were using more control deflection than required.

The P1127 aircraft in hover is considered to have an acceleration control system, and the roll control forces exceed this criterion after the first half inch of control displacement (see fig. 7). This condition has been rated satisfactory by many pilots. There are two possible explanations for this. First, data from reference 7 show that for vertical takeoffs and landings, the pilot seldom uses control displacements larger than 1.75 to 2.0 inches where the control forces exceed the criterion limit but are still low. Second, the criterion for roll control force gradients for an acceleration control system may be too low. To make the criterion compatible with the P1127 data requires that either the allowable breakout force be increased from 1.5 to 2.0 lb or the allowable gradient be increased from 1.25 to 1.5 lb/inch. At the present time, it appears that the 1.25 lb/inch force gradient limit may be too low for acceleration systems.

The XV-5A longitudinal and lateral forces (ref. 9) were considered too high for low-speed maneuvering and were assigned a pilot rating of 3. The longitudinal force for 1-inch control displacement (fig. 6) is equal to the criterion limit for a longitudinal rate system, indicating that this limit is approximately correct, at least for the first inch of travel. The lateral force (fig. 7) is below the criterion limit for a lateral rate system, indicating that this limit may be slightly high.

The XC-142 yaw control force gradients plus the breakout forces produced a total force of 70 lb/inch of pedal displacement and was assigned a pilot rating of 4. If this total force were reduced by 1/2, as recommended by the pilots, it would still exceed the total force criterion limit by 15 lb. It is interesting to note, in reference 5, that these pedal forces (70 lb/inch) were too high for hover; however, in cruise flight, the directional control was rated to be too sensitive. This illustrates the compromises and tradeoffs that can be encountered in designing a V/STOL control system.

The control force ratio (control harmony presented in table 1.4) used to establish longitudinal and directional force gradients from lateral force gradient data is not well documented and should be the subject of further simulator/flight studies. Data indicate that for hover and very low-speed operation, the control force ratio should be approximately 1. Both the P1127 aircraft and SG-1262 hover rig have longitudinal/lateral control force ratios of approximately 1 over the first 2 inches of travel (figs. 6 and 7). The data indicate also that the maximum directional/lateral ratio desired for STOL aircraft is twice the value desired for VTOL aircraft (see figs. 5 and 8 and table 1.4).

Table 1.4

Control Force Harmony Ratio Criteria

Control Force Ratio	Type of Aircraft	Minimum Ratio	Optimum Ratio	Maximum Ratio
Longitudinal / Lateral	V/STOL	1	2	4
Directional / Lateral	VTOL	4	6	8
Directional / Lateral	STOL	4	8	16

The STOL force gradient criteria presented in table 1.4 were established from the STOL data summary of figures 3-5. It should be noted that the pilot ratings were not obtained from a systematic study and refer to the complete control system of which the control force harmony ratio is only one of many important characteristics. The limited data indicate that pilots prefer a higher directional force harmony ratio for STOL aircraft than for VTOL aircraft. Also, the data indicate that a longitudinal/lateral ratio greater than 1 is desired for STOL flight.

1.4 CONTROL SYSTEM FREE PLAY

It is necessary to limit the amount of free play in the control systems for V/STOL aircraft because of the direct effect on precision of control. An exact amount of free play, or an allowable free play based on a percentage of full control travel, is not specified in section 1.4 (ref. 3) because the allowable value depends on the V/STOL concept. For example, in reference 10 it is shown that a very small amount of free play can have a severe adverse effect on control response. This V/STOL concept utilized four tilt propellers for hover control on two tandem wings. The aircraft had the following control response hysteresis bands: longitudinal, 4-8%; lateral, 12-25%; and directional, 8-14%. These results show that the lateral system had three times the hysteresis of the longitudinal system and the directional system twice that of the longitudinal. These ratios are significant in that they were approximately the same as the respective ratios of propeller blade angle change per percent of control displacement and all were equivalent to a 0.3° propeller blade angle change. Analysis of the propeller pitch change rod travel at the point of connection to the hydraulic propeller pitch valves showed that a free play of only 0.0129 inch would result in the stated hysteresis bands.

Normally, free play is checked or evaluated only about the neutral or trim control position. However, it is possible to have free play at the extremes of control travel. This can arise more frequently on V/STOL aircraft where two different moment producers are usually utilized. The documentation for control travel (section 1.9) presents several cases where free play was encountered at the extremes of control travel.

1.5 POWERED CONTROL SYSTEMS

Powered control systems, commonly used on high performance and V/STOL aircraft to meet desired low friction and force gradient values, can be limited in their usefulness by improper design. For example, low supply line volume in a hydraulic system can result in low actuator rates, resulting in nonlinear force and control deflection characteristics that are particularly bothersome for V/STOL aircraft which require rapid and frequent control inputs for precise maneuvering in gusty air.

The STOL handling qualities of the C-130 aircraft (ref. 4) were compromised because the roll and yaw control actuators were rate limited. In reference 6 (CL-84), it was reported that the wing tilt operation and landing gear retraction could not be performed simultaneously because of capacity limitations of the hydraulic system. In reference 10 (X-19), it was reported that ground checks of the secondary hydraulic system (1500 psi) revealed that a pressure drop-off occurred with landing gear extension or retraction. This 200-psi loss in pressure required 15 to 20 seconds to return to normal system pressure. This loss in pressure could have affected the flight-control system because of the interconnection of the main landing gear and pitch and roll control boost systems in the hydraulic system.

1.6 TRIM SYSTEMS

The criteria of reference 3 require that the trim systems be capable of reducing the control forces to zero for any condition where prolonged steady operation is required. Another important factor is the trim rate or the time required to trim the forces to zero. A system capable of reducing the forces to zero, but too slow in operation, can limit the operational capability of the aircraft. On the other hand, trim rates that are satisfactory for low-speed flight may be too sensitive for high-speed flight. For example, the XV-5A longitudinal trim rate was initially set at 0.2 deg/sec and was too slow at airspeeds less than 150 knots. The trim rate was changed to 0.4 deg/sec and this was too fast at airspeeds above 250 knots. For this reason a variable trim rate device was recommended (ref. 9).

As specified in the criteria, actuation of the trim system should not produce "stick jump" nor should trimming of one axis result in an out-of-trim condition in another axis. It is difficult to satisfy this criterion when a "press-to-release" system is used. In press-to-release systems, one button is actuated that zeros the pitch, roll, and yaw control forces. In fact, the main advantage of this system is the ability to trim all three axes simultaneously. Conversely, it is difficult to trim one axis without introducing some position change in the other two axes. As long as the trim button is depressed, the spring force-feel device is released from the control system and the pilot has a zero force gradient and zero or very low breakout forces. In this condition, it is very easy to introduce an unwanted trim change in another axis. For example, a pilot attempting to trim a large longitudinal force can experience longitudinal "stick jump" caused by the sudden release of the force and also have "control jump" in the other two axes that is related to the roll and yaw breakout forces. The force a pilot can hold without producing a control movement is directly related to the breakout force with the spring "force-feel" device engaged. If the roll and yaw axes have large breakout forces, considerable control motion can result when the spring force is momentarily released from the system while attempting to trim a longitudinal force.

Following a failure of any trim system, the permanent out-of-trim forces should not exceed 10 lb for pitch, 7 lb for roll, and 20 lb for yaw. Considering the maximum control travel permitted in section 1.9 and the maximum control force gradient permitted in section 1.3, the out-of-trim forces are approximately half the maximum allowable forces. More specifically, the maximum out-of-trim forces specified in the criteria permit a 3.5-inch pitch trim change, a 2.5-inch roll trim change, and a 2.5-inch yaw trim change. It is possible, within these criteria, to handle relatively large trim changes and still have control forces that the pilot can hold for an extended period of time.

Failure of a powered control system or SAS should not affect the ability to trim. Reference 5 shows that loss of roll or yaw SAS on the XC-142 aircraft would affect roll and yaw trim. Since each SAS axis had two channels, loss of one channel would reduce trim authority 50%. Loss of both channels would result in complete loss of trim. Because of SAS redundancy, complete loss of roll or yaw SAS and therefore loss of trim rarely occurred. This coupling of SAS and trim did have one advantage. A run-away trim condition could be corrected by disengaging the SAS axis. Care must be exercised here to prevent the disengagement transients from being more severe than the run-away trim condition.

1.7 HEIGHT FROM CONTROL SYSTEMS

The criteria of reference 3 state that the height control should remain fixed at all times unless moved by the pilot or some automatic system. In addition, the criteria require a zero force gradient plus an adjustable friction damper that provides a constant force at least equal to the breakout force specified in section 1.2. To meet similar criteria, some helicopter collective controls have been equipped with a magnetic brake. This magnetic brake operates very similar to the press-to-release trim device described in the previous section. When the brake is released to make a collective control change, the breakout and friction forces are reduced to zero. This makes it easy to encounter a vertical P.I.O. and clearly shows the need for some friction or breakout force to prevent unwanted control motion. Additional data on height control systems have been discussed in section 1.2.

If an aircraft is equipped with both power lever and lift-stick (collective) controls, the conversion from one system to another should be accomplished easily with a minimum of procedural complexity. Reference 5 reports an outstanding example of procedural complexity that was required to switch foom a collective lever to throttles on the XC-142 aircraft. It was as follows:

1. Disengage the collective lever from the throttles by pressing the throttle disconnect button on the collective lever grip.
2. Slowly raise the collective lever to the upper stop to position the beta linkage while the propeller governor maintains rpm.
3. Disengage the collective lever from the beta linkage by pressing the beta latch located on the collective grip.

4. Lower the collective lever to the stowed position in a detent on the deck of the cockpit.

Reference 5 recommends that the use of two different controls for power management would be undesirable for a production V/STOL aircraft and that a throttle control system should be provided for all regimes of flight.

Another height control problem was reported in reference 7 (P1127). In this case, the thrust vector control lever and power control (height control) were mounted side by side on a single quadrant. Hovering height control was achieved by means of the power lever. Power lever movements, required to maintain a stabilized height were very small, normally yielding 0.25 to 0.5% rpm change. The quadrant was equipped with an adjustable friction lock or damper whose proper adjustment prior to VTOL flight was a matter of some importance. If the friction were set too high, the throttle breakout forces precluded smooth power adjustments and a stabilized hover height was difficult to achieve. If the friction were too low, the throttle tended to creep aft when released. On one occasion, a maneuver involving nozzle movements while in a hover had to be aborted because every time the throttle was released the rpm dropped sufficiently to require immediate action to correct the developing sink rate. Being unable to release the control stick and unwilling to allow a high sink rate to develop, the pilot could not proceed with the tests without first landing and adjusting the friction. It was recommended that a minimum level of friction that would prevent throttle creep be incorporated. This requirement is already covered in section 1.2, but is repeated here to emphasize the need for a separate friction device for the power control and the thrust vector control that would permit the pilot to fix the position of one control and still move the other smoothly. The P1127 had this, but the minimum power lever friction was too low.

1.8 THRUST VECTOR CONTROLS

The criteria of reference 3 relate to the longitudinal type of thrust vector control and a minimum or maximum thrust vectoring rate is not given. The pilot desires to have direct control of the thrust vectoring rate which makes the criteria different from previous requirements. It will be shown that these are governed by the VTOL concept.

Controls criteria for thrust vectoring along the longitudinal axis must consider the accelerating and decelerating transition as well as the low-speed maneuvering about the hover condition. When maneuvering about the hover condition, the magnitude of the engine thrust vector is nearly constant, the aerodynamic vectors are negligible or small, and power management is consequently a secondary requirement. During a maximum performance accelerating transition, however, the magnitude of the engine thrust vector is again approximately constant but the aerodynamics vector varies appreciably throughout the transition. In both these cases, management of the engine thrust vector angle and aircraft attitude are the pilot's primary objectives. Several different types of thrust vector controls have been used satisfactorily in these two areas. Data presented in section 5.2 (Acceleration-Deceleration) show that the pilot used approximately the same thrust vectoring rate to achieve nearly identical acceleration characteristics regardless of the type of control.

It is the decelerating transition that determines the rate of longitudinal thrust vector control desired. At the start of the decelerating transition, the aerodynamic vector is large and the engine thrust vector is usually small. During the deceleration, these conditions reverse and the pilot must be required to manage power, thrust vector angle, and aircraft attitude. If the small engine thrust vector can be rotated without tilting the aerodynamic vector, as on the P1127, the pilot can use a high thrust vectoring rate to quickly obtain any desired position. He is then free to manage power (engine thrust vector magnitude) during the deceleration. If the engine thrust vector cannot be rotated without tilting the aerodynamic vector, as on tilt-wing aircraft, power, vector angle, and aircraft attitude must be managed simultaneously. The pilot is also limited to a much lower thrust vectoring rate which is a function of the aerodynamic magnitude and tilt rate (see section 5.2). To satisfactorily manage these variables during a decelerating transition. it is necessary to combine the thrust vector control and the power control. This can be accomplished by employing a thrust vector control button on the power control. As noted in the following, this can lead to some human-engineering problems that affect the pilot's ability to perform the task and adversely affect the pilot rating. On the CL-84 (ref. 6) the wing tilt switch (thrust vector control) was located on top of the power lever and was difficult to operate, particularly during transition where power lever movements were opposite in direction to wing-tilt switch movements. It was recommended that the tilt switch be located on the right side of the power lever in the natural left thumb position. Also, the need for a variable-rate wing tilt control switch was considered mandatory on production tilt-wing aircraft.

If a separate lever-type thrust vector control is used, as on the P1127 aircraft, the problem reported in reference 7 should be avoided. The thrust vector lever and the power lever had similar mechanical actions that could confuse the pilot and result in a dangerous or catastrophic condition. To select a 95° (5° forward of vertical) thrust vector angle, it was necessary to move the control to the 90° position and then lift the control over a stop to the 95° position. The power lever had an identical motion. To shut down the engine, it was necessary to move the power lever to an idle stop, then lift the lever and move it to the fuel-cutoff position. It was reported that on six occasions, during the Tripartite Trails, the pilots with considerable experience in the airplane inadvertently shut down the engine during landing roll when they had intended to select nozzle braking. A switch operated thrust vector control system was installed in a test configuration (reference 11) to avoid this problem.

To briefly summarize this documentation, it appears that a satisfactory thrust vector control (for any axis) should provide, (1) a pilot-controlled variable rate and (2) a minimum and maximum rate to be determined by the VTOL configuration. Also, it should be remembered that for V/STOL aircraft the thrust vector control is a primary control and a corresponding human-engineering effort should be expended.

1.9 CONTROL TRAVEL LIMITS

The control travel criteria of section 1.9 (ref. 3) specify large ranges for pitch, roll, and yaw cockpit control travel (table 1.5). Control travel for various V/STOL aircraft with comments are presented in figure 13.

Table 1.5			
Control Travel Limits			
	Longitudinal, in.	Lateral, in.	Directional, in.
Cockpit Control Travel	±4.0-±6.5	±3.0-±6.5	±2.5-±4.5

These ranges of control travel are considered necessary because of the increased capability and requirements of V/STOL aircraft control systems. For example, the lateral control may command changes in propeller blade angle and/or aileron, such as on the XC-142 and BR 941 aircraft. It is possible that maximum control travel will be different for different modes of flight to obtain optimum control sensitivity.

As stated in the criteria, full hovering control power may be obtained by less than full cockpit control travel to achieve the required sensitivity. This means that the final portion of control travel will not produce an increase in control moment, but will still move a moment-producing device such as an aileron or elevator during hovering flight. This can produce an erroneous control margin whereby the pilot believes that he has more control moment available than actually exists. The size of the effective control travel range and total control moment available will be readily apparent to the pilot, however, when full control moment is required for trim.

In the following paragraphs several examples are reviewed in which increased sensitivity was obtained by scheduling full hover control power into an effective range that is less than the normal control travel range for aerodynamic flight.

Reference 12 presents a time history (which is reproduced in fig. 13) that shows a large transport performing a lateral 200-ft sidestep maneuver during final approach. This aircraft had a total lateral wheel control travel of 75°, but total control moment was obtained in the first 30° of wheel travel. This system was rated satisfactory even though the pilot used the full 75° wheel travel for as long as 2 seconds. Two factors contributed to a satisfactory rating. First, trim requirements were low. Second, full control moment was commanded very quickly but for short periods of time so that the pilot was not aware of the fact that the last 45° of wheel travel did not produce an increase in rolling moment.

Data from reference 5 show that the XC-142 aircraft, with a wing angle of 80°, had 4.2 inches of forward longitudinal control travel, of which only the first 1.2 inches was effective. The last 3 inches did not produce any change in tail propeller blade angle. This control system also had free play as the last 2 inches of control travel did not move the unit horizontal tail or change the tail propeller blade angle. The total forward travel of 4.2 inches with only 1.2 effective inches gave the pilot the impression of having more control moment than was actually available. After displacing the control 1 inch forward, the pilot found that the trim requirements generated by the maneuver exceeded the hover control moment available. This produced an uncontrollable pitch-up condition as the remaining control travel was ineffective and the pitch rate could only be checked by reducing power. Entry into this condition would have been more difficult if the available control moment had been stretched over a larger effective control travel range. This would have required a larger control input to achieve the same aircraft response, however, the reduced control margin would have served as a warning to the pilot. The sensitivity in this case would have been unacceptably low and the overall handling qualities downgraded accordingly. It must be remembered that in this aircraft, speed changes were nominally accomplished by changing wing angle. When speed changes were made through pitch attitude changes, the automatic trim capability would be bypassed since this was programmed as a function of wing angle.

The SG-1262 hover rig had ±4.0 inches of control travel for each axis, but the effective range was different for each control. The pitch control had an effective range of ±2 inches with bleed air only and ±2.5 inches with thrust modulation; the lateral control had an effective range of ±1.0 inch; 75% of yaw control was obtained in the first 0.75 inch of travel with the remaining 25% obtained in the last 3.25 inches of travel. This VTOL vehicle was flown by seven pilots and available data do not indicate any criticism of these effective control ranges.

In summary, control travel for V/STOL aircraft is a variable that must be tailored to the V/STOL concept and flight condition. It may be permissible and satisfactory under certain conditions to have different control travel limits and different effective ranges for various phases of the flight regimes.

1.10 CHARACTERISTICS OF AUGMENTATION SYSTEMS

The criteria of section 1.10 (ref. 3) state that stability or control augmentation systems that are employed to improve one part of the aircraft behavior should not have an adverse effect on other handling qualities. Probably the best example of this is the addition of attitude and heading hold features to improve hovering stability which results in decreased low-speed maneuverability of the aircraft. In references 5 and 6, the pilots of the XC-142 and CL-84 stated that with attitude or heading hold engaged the aircraft was too stiff and they preferred to maneuver by reverting to a rate-damped system; i.e., they were willing to give up some stability for more maneuverability.

Another possible undesirable feature of SAS is a nonlinear response that can occur when the SAS saturates. On the CL-84 (ref. 6), which employed an augmented yaw rate damping, the pilot commented that the aircraft was very stiff directionally until the SAS saturated, after which an apparent rapid increase in yaw control sensitivity occurred.

The ability to switch smoothly from an attitude and/or heading hold system to a rate-damped system is a desirable feature. Systems that depend on yaw/angle-of-attack vanes and/or airspeed measurements to perform switching functions can be subject to erratic operation. For example, an aircraft system that depends on airspeed measurement to switch from yaw rate damping to heading hold will perform erratically unless low airspeeds can be measured accurately.

Some SAS-related problems can be misleading and the solution may be only indirectly related to the SAS system. Unpublished data from the VFW SG-1262 hover rig described a problem of this kind. Large-amplitude, pilot-induced roll oscillations (±30°) were experienced during the early development flights of this vehicle even though an attitude control system was employed. The roll attitude control system permitted a maximum bank angle of 12° for full control deflection. The PIO problem was encountered when the pilot translated sidewards at speeds greater than 20 knots and then abruptly (less than 0.5 second) commanded maximum bank angle in the opposite direction. Momentum drag of the vertically mounted engines produced a rolling moment that was opposite to the initial bank angle commanded and proportional to sideward velocity. When opposite bank angle was suddenly commanded, at speeds greater than 20 knots, the resultant roll rate generated by the engine inlet momentum drag plus the control reversal input was greater than the SAS could handle. As a result, the vehicle overshot the maximum 12° bank angle, causing the pilot to again reverse the control input and a pilot-induced oscillation developed. Exit from this condition was accomplished by releasing the cockpit roll control which the pilots were reluctant to do. The solution was to increase roll control power so that a greater control margin was available to the SAS to counter the large roll rates that could be developed.

Insufficient data are available to completely define SAS failure criteria, except to say that failures shall not result in objectionable out-of-trim moments or other unsafe flight conditions. Indication of SAS failure, however, should be immediately apparent to the pilot in such a manner that he can identify the system and extent of failure. Reference 5 defines a dual-channel SAS in which a failure in one system will cause both channels to lock the controls in a neutral position. When failure occurred, the pilot was required to turn on one channel at a time to determine which channel had failed.

In addition to immediate indication of failure, SAS should provide protection against;

(a) large pendulum-type oscillations that can result from failure of a rate feedback circuit while the attitude feedback circuit is still operating,

(b) large amplitude control inputs that can occur when an attitude or other high-gain system attempts to maintain a given attitude or a given response during the ground roll. (Feedback circuits should be modified when the aircraft weight is on the landing gear to minimize this problem.)

Finally, additional data and experience are required before criteria can be written that will cover all types of SAS, including the more recently developed velocity command systems.

Section 2.0

LONGITUDINAL STABILITY AND CONTROL

2.1 GENERAL

Good longitudinal stability and control characteristics are essential if V/STOL aircraft are to operate routinely into and out of confined areas. In general, longitudinal stability, damping, and control deteriorate at low speeds and the combined effects can result in poorer precision in flight-path tracking.

Factors which individually influence the longitudinal behavior of conventional aircraft have been studied for several years; Mil Spec 8785B has detailed requirements covering the speed range down to stall. In addition to meeting the foregoing requirements, V/STOL aircraft must also fly down to hover. In this lower speed regime, there is less information upon which to base requirements and many factors must be considered individually and in combination for setting up meaningful criteria. Some of the characteristics which strongly influence the longitudinal behavior include: control power and sensitivity, linearity of response, pitch damping, control-system time constant, control forces, cross-coupling, normal acceleration sensitivity, flight-path speed stability (backside operation), angle-of-attack stability, lift-drag variation with engine power, effects of proximity to ground, and direct lift and/or drag control.

2.2 PITCH CONTROL POWER

The pitch control power criteria of reference 3 is presented in table 2.1

Table 2.1

Pitch Control Power Characteristics

Parameter to be measured	Control power required for	Type of control system	Minimum levels for satisfactory operation: Hover	STOL
Pitch angular acceleration, rad/sec^2	Maneuvering	Attitude command	0.1-0.3	0.05-0.2
		Rate	0.1-0.3	
		Acceleration	0.2-0.4	
Pitch angle after 1 sec, deg	Maneuvering	Attitude command		
		Rate	2-4	2-4
		Acceleration	2-4	2-4
Pitch control deflection at zero pitching velocity in.	Trim	All	Sufficient control in EXCESS of MANEUVERING REQUIREMENTS to trim over designated speed and c.g. range and for most critical engine failure	
Time to recover to initial attitude, sec or control deflection, in.	Upset (due to gusts, recirculation, ground effect, etc.)	All	Sufficient control in EXCESS of MANEUVERING and TRIM REQUIREMENTS to balance moments due to a specific gust; e.g., 30 ft/sec gust:	
			Building up in 1 sec	Building over a 100-ft distance
Pitch control power angular acceleration, rad/sec^2	Typical range of values used by V/STOL aircraft for maneuvering, trim, and upset	All	0.4-0.8	0.4-0.6

The total amount of pitch control power desired by the pilot for VTOL and STOL operation is determined primarily by three individual requirements: (1) how rapidly the aircraft must be maneuvered for a particular task; (2) the magnitude of the moments that must be trimmed out to maintain a given attitude or speed when power, flaps, or thrust vector are changed; and (3) the amount of pitch attitude change required to compensate for gusts, recirculation, or other disturbances.

2.2.1 Control Required for Maneuvering. The pitch maneuvering control power criteria are presented as a range of values in which minimum angular acceleration and/or a minimum pitch attitude change after 1 second are listed for hover and STOL operation for three types of control systems. These values were established by reviewing the available flight-test data, simulator studies, etc. (a tabulation of data is given in figs. 14-15) and considering the following:

(a) Pitch control power actually used by V/STOL aircraft in maneuvering
(b) Control travel limitations
(c) Minimum pitch control sensitivity requirements
(d) Minimum pitch damping requirements
(e) Use of thrust vectoring and/or attitude changes

The values listed in table 2.1 for maneuvering may appear to be low based on the total control power available; for example, higher values are listed in the studies of reference 13. It must be emphasized, however, that these values are minimum pitch control power requirements only for maneuvering. The lower value is a minimum for aircraft which can use thrust vectoring to augment pitch attitude changes for longitudinal positioning while the upper value is a minimum for aircraft whose mission requirements are satisfied by minor pitch attitude response. Substantiation of these values is given by the data in figure 16 in which comparisons are made between simulator and flight based on calculated responses using pitch control sensitivities and optimum pitch damping of several VTOL aircraft.

Data from reference 14 indicate that pitch maneuvering control powers lower than these minima may prove satisfactory. Figure 17a presents average linear mean values of pitch control power usage for different tasks performed by the VJ-101 aircraft and hover rig. Figure 17b presents maximum pitch control power values covering 99 percent of total usage for these same tasks. The average linear mean values are considerably lower than the minimum maneuvering control power criteria of table 2.1, while the maximum values used fall in the center of the criteria range. In general, low response values were satisfactory for this fighter VTOL aircraft since landing and takeoff were considered only incidental to the overall mission for this class aircraft.

The minimum pitch maneuvering control power criteria for STOL aircraft (table 2.1) were established chiefly from data contained in reference 4. The eight STOL aircraft documented in reference 4 used very little pitch maneuver control power during climbout after takeoff or during the landing approach. Changes in the flight-path angle, especially during landing approach, were accomplished in part by changes in thrust (powered lift). The major requirements for pitch maneuvering control power occurred during the rotation to takeoff attitude and during the flare to a landing attitude.

More operational experience is needed to more accurately define desirable maneuvering responses for various missions since it is recognized that the minimum values listed may not adequately reflect the needs of all V/STOL operations.

2.2.2 Control Required for Trim. The pitch trim control power criteria presented in table 2.1 reflect the need to consider aircraft concept. For example, for VTOL aircraft with thrust vectoring capability, the pitching moments that accompany a change in airspeed are usually less where thrust vector angle is used. This indirect method assures that for a given reference pitch attitude (usually zero deck angle), a given thrust vector angle will produce a specific trim speed. Larger out-of-trim conditions can result if the pilot commands speed changes through pitch attitude with a constant vector angle. The size of these trim changes and the pilot's ability to precisely compensate for them determines the width of the transition corridor.

It is very possible that the trim requirements may be the dominant factor in determining total control power requirements for VTOL aircraft when operating out of ground effect (OGE). If there is insufficient control power available to compensate for trim changes, maneuverability of the aircraft will be limited. Examples of these limitations are presented in the following paragraphs.

The XC-142 longitudinal control system was designed considering the following: (1) speed changes would be commanded by changing wing angle and (2) pitch trim changes would be programmed as a function of wing angle. If, however, the pilot held the wing angle constant and commanded a change in speed by changing pitch attitude, he could bypass the programmed trim changes. This resulted in a long period (20 seconds) oscillation in pitch attitude which was attributed to propeller pitching moments arising from changes in translational velocity. This was reported in reference 5 as follows: "As the nose was pitching up due to propeller moments, full forward stick had little effect on the pitching. This oscillation was never permitted to go more than one terrifying cycle, so it was not determined whether it was damped or divergent or what effect changes in power would cause."

The unique thrust vectoring system used on the X-14 aircraft also created large pitch trim changes that were greater than the hover pitch control power (ref. 15). For this reason, thrust vectoring could not be initiated from hover. This limited the flexibility of the aircraft and it was necessary to accelerate from hover by changing pitch attitude. This is the reverse of the XC-142 problem previously described where it was necessary to use thrust vectoring to avoid the out-of-trim condition.

Data from reference 6 show that the CL-84 aircraft with a wing angle of 40° (STOL operation) experienced a large nose-down trim change in ground effect (approximately 10 ft wheel height) that required 90 percent of the available aft control travel to maintain a level attitude for landing. This trim change was rated mild by the pilots because it occurred at a point during the landing where the pilot would normally be flaring for touchdown.

A final example shows how reducing the trim requirements improved the flexibility of the VZ-2 aircraft. Data contained in reference 16 show that the addition of a full-span flap to the VZ-2 tilt wing aircraft reduced trim requirements by approximately 20 percent. This provided more pitch control power for maneuvering and to compensate for gusts and other disturbances.

For STOL aircraft (ref. 4), the longitudinal control power requirements may be dominated by either maneuvering or trimming, depending on the aircraft configuration and the nature of the approach and landing technique used. The criteria for STOL pitch trim control power stated in reference 4 require that the longitudinal control be capable of trimming the aircraft throughout the flight envelope, and it must be possible to obtain the defined minimum speed at all weight and loading conditions.

2.2.3 Control Required for Gusts. The criterion presented in table 2.1 for pitch control power required to compensate for upsets (due to gusts, recirculation, ground effect, etc.) simply states that sufficient control in excess of maneuvering and trim requirements should be available to balance moments due to a specific gust. This, like the criteria for pitch trim control power, is configuration oriented

and insufficient data exist to specify a firm value. The example gust is presented solely as a guide from which more applicable values can be developed. The procuring agency must determine a value for the particular concept under consideration.

When operating in ground effect (IGE), this requirement may have a dominant effect on total control power requirements since it is very difficult to establish a trimmed condition while operating in, or performing takeoff and landings through the ground effect region and maneuvering is usually a minimum. The pitch trim control power requirements, therefore, are hidden and the requirements for pitch control power to compensate for upsets become more important. It will be shown in section 3 that the control power required to compensate for upsets about the roll axis is a critical factor in determining the total roll control power required.

Total pitch control power is presented in table 2.1 as a typical range of values used by V/STOL aircraft for maneuvering, trim, and upset (see figs. 14 and 15). Each of these aircraft had individual characteristics that dominated total pitch control power requirements. In part, the pitch maneuvering control power required depends on how much inherent rate damping and/or attitude stiffness must be overcome. A high level of damping and attitude stabilization can reduce the trim and upset control power requirements by keeping unwanted excursions to smaller amplitudes. The total control power requirements, therefore, may be less for a stability augmented aircraft.

Total pitch control power requirements can be reduced by restricting the flight conditions or maneuvers that produce high trim requirements. As discussed previously, the X-14 aircraft had insufficient pitch control power to compensate for nose-up pitching moments generated when thrust vectoring was initiated from hover. It was possible to avoid this high trim requirement by using pitch attitude change to accelerate from the hover. In this way, additional aerodynamic control power was obtained that would permit thrust vectoring to be used. In addition to this problem, two other flight conditions resulted in nose-up pitching moments that overtaxed the X-14 pitch control system: (1) high vertical climb or descent rates and (2) high rearward velocity translations. Both conditions were experienced while attempting to hover at altitude (2500 ft) where the pilot had poor hover references. He was not aware of the large vertical or rearward velocities until the nose-up pitching occurred. By decreasing the operational flexibility of the aircraft, the flight conditions that produced high trim requirements were avoided and total control power requirements were reduced. Some flight conditions, however, could not be avoided, such as the control power required to compensate for disturbances experienced while operating IGE When the X-14 had sufficient control power to compensate for disturbances IGE, it had sufficient control power for normal operation OGE.

An interesting comparison between two aircraft using the same VTOL concept but differing in size is presented to illustrate the range of pitch control power required. The XC-142, a four-engine tilt-wing aircraft (37,000 lb), had a total pitch control power of ± 0.45 rad/sec^2 while the CL-84, a two-engine tilt-wing aircraft (10,600 lb), had a total pitch control power of ± 1.35 rad/sec^2. As discussed previously, the XC-142 had insufficient control power to trim the nose-up pitching moments generated by large translational maneuvers (0-30 knots) that were performed by changing pitch attitude with a fixed wing angle. The CL-84, with its higher control power, appeared able to operate in is area without experiencing any control power limitation. The pilots of both aircraft commented that small translations were easier to perform using pitch attitude changes when the attitude stabilization was turned off. With it on, the aircraft were too stiff in pitch and this required additional control input to achieve the desired response. It must be remembered that both aircraft were designed to maneuver longitudinally, by using wing tilt (thrust vectoring), and maneuvering by pitch attitude changes was a secondary, although desirable, feature that increased the aircraft's operational flexibility. The CL-84, in the STOL approach configuration (40 knots), encountered large nose-down pitching moments in ground effect that required 90 percent of the available nose-up control power to trim. The XC-142, however, could not operate in this flight condition because of large lateral-directional disturbances (see section 3).

The foregoing was presented to show the range of control power required for various V/STOL concepts and the effect on the aircraft's operational capabilities. Both aircraft had satisfactory pilot ratings when flown as designed, that is, when wing tilt was used for longitudinal maneuvering.

Figure 17 presents control power usage data from reference 14. The inference is that the hover-rig provided data that were directly applicable to the VJ-101 aircraft and demonstrated the usefulness of hover rigs. Data in figure 17 also show that the maximum control power values covering 99 percent of total usage by the VJ-101 aircraft were considerably lower than the criteria of table 2.1. This is reinforced by data in reference 7 that show approximately the same control power usage for the P1127 during vertical and short takeoffs and landing. Highest control power usage for both aircraft occurred during short, roll-on landings. This probably results from the fact that the aircraft operate in ground effect for a longer period of time when performing roll-on landings. It must be remembered, however, that even though the P1127 and the VJ-101 control power usage in the test environment was lower than the criteria minima, both aircraft had installed or available control power much higher than the minimum. This high available control power gave the pilot confidence to explore operation in gusty air, and flexibility to maneuver over a wider flight envelope.

2.3 PITCH CONTROL SENSITIVITY

The pitch control sensitivity criteria of reference 3 is presented in table 2.2.

Table 2.2

Pitch Control Sensitivity				
Item	Parameter to be measured	Type of control system	Minimum levels for satisfactory operation	
			Hover	STOL
Sensitivity	Attitude change per unit control deflection, deg/in.	Attitude command	3.5	
	Pitch angular acceleration per unit control deflection, rad/sec²/in.	Rate	0.06-0.1	0.08-0.12
	Pitch angular acceleration per unit control deflection, rad/sec²/in.	Acceleration	0.08-0.16	
Linearity	Variation of pitch angular acceleration with control deflection	All	Constant or should not abruptly increase or change sign.	

The above criteria were established by considering the following: (1) total control power available, (2) control travel limits, (3) control stick gearing, (4) type of control system, and (5) the primary method used for longitudinal translation. Considering the first three items, it must be realized that the ratio of maximum control deflection to total control power available may not provide an optimum control sensitivity, but rather defines a lower limit using linear control gearing.

Numerous studies have been conducted to define satisfactory longitudinal response characteristics of V/STOL aircraft. Most of these studies have assumed that aircraft pitch attitude change was the primary method for longitudinal maneuvering, and the two primary parameters examined were control sensitivity and angular rate damping. Typical results of these types of studies are shown in figures 18 and 19 (refs. 17 and 18). The curves of constant pilot rating for different values of pitch sensitivity and damping from the individual studies show that a wide range of sensitivity and damping is acceptable. For example in figure 18, the optimum point on the Princeton 3.5 pilot rating curve produces the same response (pitch angle after 1 second) as the optimum point for the Bell 3.0 pilot rating curve. Of course, other factors such as X_u, M_u, and M_α must be considered to define satisfactory pitch behavior. For example, the Princeton aircraft that inherently had a large M_u value was disturbed more in gusty air, and a higher value of rate damping was needed to limit pitch excursions.

Low pitch sensitivity combined with high damping will cause the pilot to downgrade the control system because of low response. Decreasing damping, if this is possible, will not always solve the problem even though the pilot is now able to get the response he desires because of the tendency to overshoot the desired response. Low sensitivity is also undesirable because the control displacement required to get the desired response is too large. Large control displacements make it more difficult for the pilot to accurately return the control to the trim position.

As previously mentioned, other parameters such as X_u/m and $M_u g/I_y$, also have a strong influence on sensitivity and damping levels. The effect of these two stability derivatives, (discussed in more detail in ref. 17) will vary depending on the V/STOL concept. The stability derivative X_u/m (a measure of the change in longitudinal force with change in airspeed) serves both as translational damping when performing longitudinal maneuvers and as a means by which gusts impart longitudinal accelerations to the aircraft. As X_u/m becomes more negative (increasing drag), less control sensitivity is required, especially if an increase in speed produces a nose-up pitching moment that helps to damp longitudinal translation. At values of X_u/m near zero, relatively high levels of control sensitivity are required because of the lack of translation damping. Once the aircraft is set in motion, either by maneuvering or as a result of a gust, the aircraft has a tendency to continue translating until arrested by the pilot. Therefore, higher values of control sensitivity are primarily required at low values of X_u/m for precision stops.

The stability derivative $M_u g/I_y$ is a measure of the change in pitching moment on the aircraft caused by changes in airspeed. As $M_u g/I_y$ becomes large, control sensitivity must be increased to compensate for the increased pitching response of the aircraft to gusts, and larger excursions in longitudinal control position are required for trim during maneuvers and for long period components of a gust. For combinations of high level turbulence and large $M_u g/I_y$, high levels of damping are desirable to reduce the short period response due to gusts. An increase in damping to reduce the pitch excursions will cause sluggish pitch response when maneuvering, requiring an increase in control sensitivity.

The foregoing factors and data (figs. 18 and 19) were used to establish the upper range of minimum sensitivity levels of 0.1 rad/sec²/in. for a rate system and 0.16 rad/sec²/in. for an acceleration system. The lower minimum sensitivity levels of 0.06 and 0.08 rad/sec²/in. were established for aircraft that use thrust vectoring for maneuvering and have rate systems and acceleration systems, respectively.

The importance of pitch control sensitivity decreases when thrust vectoring is used for longitudinal maneuvering. In this case, attitude change may be used only to assist the initiation or termination of a maneuver and/or to hold a constant pitch attitude during the maneuver. The same forces act on an aircraft regardless of how the maneuver is initiated, but they may be different in magnitude and are compensated for in a different manner. The use of thrust vectoring compensates for X_u/m effects but pitch attitude control is still required for M_ug/I_y. However, the influence on pilot control is less because it is programmed out as a function of thrust vector angle. A lower sensitivity, therefore, is permissible because the demand on the pitch control has been reduced.

The 3.5°/in. sensitivity criteria for aircraft with attitude stabilized systems was established to assure at least a ±12° attitude change capability (3°/in. x 4-in. travel) when the minimum control travel criterion was used. Data from the SG 1262 hover rig have shown that ±15° of pitch attitude change was satisfactory. Data from other flight-test reports (ref. 5) show that pilots would terminate maneuvers to obtain a static longitudinal stability data point whenever the aircraft attitude approached 10-12° nose-up even though the aircraft was capable of greater steady-state pitch attitudes.

Nonlinear control systems can improve control sensitivity if two precautions are observed. First, the sensitivity must not abruptly increase with increasing control displacement as this may cause the pilot to overcontrol and/or enter into a pilot-induced oscillation (ref. 4). Secondly, the nonlinearity should not significantly conceal from the pilot the true control power remaining (ref. 5). That is, the control system should not have a large ineffective area that would lead the pilot to believe more control response was available.

2.4 PITCH ANGULAR VELOCITY DAMPING

The pitch angular velocity damping criteria of part 1 is presented in table 2.3.

Table 2.3

Pitch Angular Velocity Damping

Parameter to be measured	Type of control system	Minimum levels for satisfactory operation	
		Hover	STOL
Angular velocity damping, one/sec	Attitude command	-2.0	-1.0
Damping ratio	Attitude command	<15 percent overshoot	
Angular velocity damping, one/sec	Rate	-0.5 to -2.0	-1.0
Numbers of control reversals required to stabilize	Rate	---	Not more than 1

The effect of pitch rate damping on pilot opinion for rate systems has been discussed in sections 2.2 and 2.3. This section, therefore will discuss only the attitude systems.

An attitude control system is characterized by a second-order equation in which the damping ratio, ζ, is defined as: $\zeta = (M_q/I_y)/2\omega_n$ where M_q/I_y = pitch angular velocity damping (1/sec) and ω_n = natural frequency, 1/sec.

It is apparent that pitch rate damping must increase as the natural frequency increases to maintain a satisfactory damping ratio. For example, if the minimum pitch rate damping of table 2.3 (-2.0 1/sec) were available with a system that had a natural frequency of 4.0, the resulting damping ratio would be 0.25. A damping ratio this low would permit a large attitude overshoot and require several cycles to damp. For this reason, the second criterion that specifies that the overshoot must be less than 15 percent of the commanded attitude change was included in table 2.3. This indirectly specifies a minimum pitch rate damping that will produce a satisfactory damping ratio. In addition, the aircraft response to a control step should reach 95 percent of the final attitude in a time between 1 and 1.5 second (ref. 8). If 95 percent of the final response is obtained in less than 1.0 second, the aircraft will be considered too responsive; if longer than 1.5 seconds, the aircraft will appear to be sluggish.

2.5 CONTROL-SYSTEM TIME LAGS

The criteria of part 1 state that following a small abrupt step displacement of the cockpit pitch control, the angular acceleration response should be in the commanded direction within 0.1 second, and the time required to reach 63 percent of the initial maximum angular acceleration should meet the values shown in table 2.4.

Table 2.4			
Pitch Control Lags			
Parameter to be measured	Type of control system	Levels for satisfactory operation	
		Hover	STOL
Time from control input to 63 percent of peak angular acceleration, sec	Attitude command	<0.2	<0.3
	Rate	<0.2	<0.3
Time to 90 percent of demanded attitude change, sec	Attitude command	<1 and <2	

The values of pitch control lags shown in table 2.4 were obtained primarily from a study by the National Aeronautical Establishment of Canada using a variable stability helicopter that employs the "model-controlled" method of simulation. In this work, various values of transport time delay for five combinations of longitudinal control sensitivity and damping were examined and the results are shown in figure 20. The flight tasks included precision hovering, low-speed maneuvering, and an VFR approach (5° glide slope). Separate assessments were made by the pilots of the characteristics in the hovering and landing approach phases of the evaluation with the model being changed between the two to prevent an interaction of assessment. The same tasks were used to evaluate the influence of first-order time delays using only models A, B, and D of figure 20. In both cases, the lag was inserted between the pilot's control input and the model that calculated the pitch angular velocity. Hence, the inherent time lag of the simulated system (somewhere between a transport and first-order lag) was approximately 0.1 second and was always included. The numbers referenced in the results are what was added during the tests.

As shown in figures 21 and 22, the additional transport time lag is not detrimental until it exceeds approximately 0.2 second, provided moderate control sensitivity and damping are used. An increase in transport lag is immediately detected by the pilot when high sensitivity or zero damping is used. In the first-order lag results (figs. 23 and 24), only the zero damping case shows this immediate effect on pilot rating of increasing lag. This is reasonable in that the zero damping case is actually an acceleration control which, by itself, creates a high pilot workload. Any time lag adds to the pilot's difficulties in making precise maneuvers.

2.6 STATIC LONGITUDINAL STABILITY CHARACTERISTICS

Previous longitudinal stability characteristics presented in AGARD TR 408 did not consider the unique feature of present-day V/STOL aircraft, that is, the ability to change speed independent of aircraft attitude by changes in thrust vector angle. For this reason, the revised criteria of part 1 (section 2.6) include trim-speed stability, stability with respect to speed (static longitudinal stability), and thrust vector stability.

2.6.1 Trim-Speed Stability. This is defined as the change in trimmed control position with a change in trimmed airspeed and is sometimes referred to as "apparent-speed stability." The criteria of part 1 require that with the pitch cockpit control forces continuously trimmed to zero at airspeeds covering the range from V_{con} to hover, the curve of pitch control position versus airspeed should be stable, that is, rearward control deflection associated with decreasing trim airspeed and vice versa. These requirements should be satisfied in level, descending, and climbing flight. Examples of trim speed stability are shown in figure 25. Each point in the examples represent a trimmed flight condition. Data from reference 9 (fig. 26) show that the XV-5A aircraft in level flight, at constant angle of attack, had a trim-speed stability curve similar to the nonlinear examples in figure 25. The curve was stable from hover to 50 knots, neutral, and then unstable as speed was increased from 50 to 95 knots.

This nonlinear and negative trim speed stability was not objectionable to the pilot because the stick reversal was gradual and small in magnitude. It should be noted, however, that the XV-5A aircraft had limited acceleration capability (see section 5.2). If airspeed changes could have been made more rapidly, the unstable portion of the trim-speed stability curve could have produced a handling qualities problem, especially under IFR conditions. Further, the reversal in trim-speed stability could be more objectionable to the pilot than the unstability because of the need to constantly monitor changes in airspeed.

2.6.2 Stability with Respect to Speed. This is also referred to as static longitudinal stability and is defined as the change in control position with a change in airspeed ($d\delta/dv$, stick-fixed stability) and/or as the change in control force with a change in airspeed (df_e/dv, stick-free stability). The criteria of section 2.6.2, part 1, require that with the pitch trim, thrust vector, throttle, or selective controls at the trim setting, the variation of pitch control position and force with airspeed should be in a stable direction over a range of approximately ± 10 knots about the trim speed. If speed stability is obtained by means of SAS, SAS failure should result in only mild instability (more than 5 seconds for divergence to double amplitude). In addition, the pilot should be made aware of any unstable variation in pitch control as a warning of the possibility of insufficient control for recovery.

Documentation data for these criteria were obtained from references 5, 6, 9, and 19, the XC-142, CL-84, XV-5A, and X-22 aircraft, respectively. The need for positive speed stability and its relationship with angle of attack stability is discussed in reference 20. These results show that unstable values of speed stability caused a rapid deterioration in handling qualities making attitude control very difficult. All of the foregoing aircraft were designed to be able to change speed by changing the thrust vector angle in which

case the pitch control was to maintain a desired pitch attitude. The pilots, therefore, comment on attitude stability as well as stability with respect to speed. Attitude stability is defined as the change in pitch attitude with control displacement.

In reference 19 (X-22A), it is stated that the static longitudinal "attitude" stability was negative for all duct angles below 60° (airspeeds above 53 knots). As expected, the "conventional" measure of stability with respect to speed, given by variation of control force and position with airspeed, was also negative (see fig. 27). The variation of stick position was generally linear throughout the range of attitude changes; hence the amount of deviation from trim attitude had negligible effect on the degree of stability. Longitudinal stick force and stick position variations were significant in their lack of cues to the pilot. The maximum stick force variation was 1-3/4 lb for a 12° attitude change and 25-knot airspeed change. Stick position variation averaged approximately 1/2 in. for the same conditions. Reference 19 goes on to state that "the negative attitude stability caused difficulty in trimming the aircraft, and increasing aft stick position with increasing nose down pitch attitude was disconcerting to the pilot." This deficiency is unacceptable for IFR flight operations (pilot rating of 7) and undesirable for VFR flight (pilot rating of 4). As shown in figure 27, the static longitudinal attitude stability characteristics at duct angles between 60° and 90° (0 to 53 knots) were satisfactory (pilot rating of 3). A summary plot of the X-22 test results is presented in figure 28.

Data from reference 9 indicate that the XV-5A aircraft had negative stability with respect to speed as shown in figures 29 and 30. The negative position and force gradients existed at airspeeds greater than 50 knots and reached a maximum negative value at approximately 75 knots. This negative stability required more careful piloting techniques to avoid undesirable airspeed changes during aircraft configuration changes. In addition, the XV-5A aircraft also had negative attitude stability in that increased aft stick was required to hold an increased nose-down pitch attitude (fig. 29). Although these stability characteristics would not satisfy the stability with respect to speed criteria of section 2.6.2 (ref. 3), they were given a pilot rating of 3-1/2. Also, the pilots did not comment directly on the negative attitude stability characteristics. A possible reason for this is that the XV-5A aircraft had a relatively flat power-required curve whereby small changes in attitude from the desired trim attitude did not produce appreciable changes in vertical speed. Further, variations in attitude resulted in only small deviations from the desired trim speed and therefore did not produce distracting vertical velocities as would result if the power-required curve were steeper. A summary of the XV-5A characteristics is presented in figure 31.

The speed stability of the CL-84 aircraft (ref. 6) was examined for two modes: (1) the airspeed was controlled by changing wing angle while the fuselage was held level and (2) airspeed was controlled by varying fuselage attitude with a constant wing angle. Controlling airspeed through this secondary mode was undesirable because of the inverse relationship between speed and power at intermediate wing angles (steep power-required curve). A rate of climb accompanied a nose-down attitude change and a rate of descent accompanied a nose-up attitude change. With SAS on, figures 32 through 35 show weak negative stability at 30° wing angle (pilot rating of 3) and a weak positive stick position stability. With SAS off (fig. 33), a neutral to negative stick position stability was noted (pilot rating of 4). This characteristic was not objectionable, but it was closely related to and dependent on attitude stability. The large positive attitude changes for moderate speed changes (knots/deg) were a desirable characteristic (pilot rating of 1). However, the weak to negative attitude stability (fig. 34) (inches of control displacement per degree of attitude change) was objectionable on conversions and reconversions, and it made stabilization of the aircraft on a given attitude difficult at fixed conversion angles. Pilot effort increased markedly with decreased visual cues, which occurred when operating at high attitudes. The pilot ratings of CL-84 attitude stability (SAS on) for different wing angles are listed below in table 2.5. Large pilot effort was required to hold attitude under the SAS off conditions (fig. 35). This was very objectionable and the attitude holding task for VFR conditions was given a pilot rating of 6.

Table 2.5

Pilot Ratings for Attitude Stability

Wing Angle	Pilot Rating
15	4
25	4
30	5
40	5
50	4
55	3

The XC-142 static longitudinal stability was evaluated with wing/flap angles of 0°/30° to 35°/60° (ref. 5) and results are presented in table 2.6 and figures 36 and 37.

Table 2.6				
	Static Longitudinal Stability			
	Control force stability		Control position stability	
CA[1]	Below trim	Above trim	Below trim	Above trim
0/30	Neutral	Neutral	Slightly positive	Slightly positive
5/30	Neutral	Neutral	Slightly positive	Slightly positive
10/30	Neutral	Neutral	Slightly positive	Slightly positive
25/30	Slightly positive	Slightly positive	Slightly positive	Slightly positive
25/60	Negative	Positive	Negative	Positive
30/30	Slightly positive	Slightly positive	Neutral	Positive
30/60	Negative	Positive	Negative	Positive
35/60	Negative	Negative	Negative	Negative
[1]Tail propeller engaged.				

For these tests (wing/flap angles of 25°/30° to 35°/60°), the fuselage was trimmed level and then pitch attitude was changed in 5° increments. This technique was used because the aircraft was flown fuselage level in the powered-lift regime, and pilots used pitch attitude instead of airspeed as a primary flight reference. The neutral and negative longitudinal control force and control position stability did not receive adverse comments from the pilots, although it was acknowledged that positive stability would be highly desirable to ease the task of controlling airspeed and/or pitch attitude. Flight at a wing/flap angle of 35°/60° was very demanding of the pilot. Flight under IFR conditions would increase the demand for positive control force and control position stability.

The foregoing documentation was presented to show that positive static stability with respect to speed is desirable, although the pilots' emphasis shifts from speed stability to attitude stability as powered-lift increases.

2.6.3 Thrust Vector Stability. The criteria of section 2.6.3 (ref. 3) require that for a given fuselage attitude and power held constant in level, descending, and climbing flight, the variation of airspeed with thrust vector position should be in the stable sense; that is, decreases in aft thrust vector angle from the vertical should result in increased airspeed and vice versa.

In general, all V/STOL aircraft satisfy these criteria except the X-14 aircraft, which has an unconventional thrust vector rotation. The rotating-cascade diverter system can rotate the engine thrust from 90° (hover position) to the cruise position of 22° from the horizontal. However, this rotation is not confined to the X-Z plane as initial thrust vector motion takes place in the Y-Z plane (laterally). This lateral thrust vector motion results in a loss in lift without a corresponding increase in longitudinal thrust. This neutral thrust vector stability has been noted by pilots to be undesirable.

2.7 LONGITUDINAL CONTROL CHARACTERISTICS IN MANEUVERING FLIGHT

The criteria of section 2.7 (ref. 3) require that at the most critical load in turning flight, at constant airspeed, or in turns with varying airspeed, increased pull forces and aft motion of the cockpit control should be required to maintain increases in normal acceleration, angle of attack, or nose-up pitch rate and vice versa.

Data pertaining to longitudinal characteristics in maneuvering flight were obtained from references 5, 6, and 19 for the XC-142, CL-84, and X-22 aircraft, respectively. It is recognized that V/STOL aircraft have a limited g capability in the low-speed flight regime, and pilots consider stick force per g a low level of importance. In references 6 and 19, the pilots rated control response (pitch rate per inch of control displacement) as more important than stick force per g gradient for low-speed maneuvering. However, this criterion does guard against an undesirable condition that existed on the XC-142 aircraft. During low-speed maneuvering (35 knots), this aircraft had a positive stick force per g gradient and negative position gradient (fig. 38). That is, to increase g level, an increased aft stick force was required, but a forward movement of control position was needed. This can lead to a dangerous situation because the aft stick force will erroneously indicate to the pilot that he is compensating for a nose-down pitching moment when in fact, the control could contact the forward stop, causing the aircraft to pitch up uncontrollably. The reason for this negative position gradient was attributed to faulty SAS operation and, in this case, was not severe enough to require full forward control. At higher speeds (135 knots), the XC-142 aircraft had a linear gradient up to approximately 1.7 g as shown in figure 39. At the higher g values, a flattening of the stick force per g gradient is evident. The response of the aircraft in this configuration (0° wing angle, 30° flap angle) was described by the pilots as crisp. No buffet was noted up to the maximum load factor of 2.35 g that required 3/4 of the allowable stick movement. Under the calm test day conditions, this configuration was given a pilot rating of 2 for traffic pattern and instrument approach conditions. It is of interest to note that the nonlinear stick force gradient did not degrade the pilot rating.

As shown in figure 40, the CL-84 control position gradient with normal acceleration at 100 knots (15° wing angle, 40° bank angle) was essentially linear (up to a limit of 1.3 g). At 57 knots (30° wing

angle, 30° bank angle) the same stick position versus load factor relationship was indicated (fig. 40), but the small load factor obtainable limited the usefulness of the data. For this reason, load factor was not considered a criterion for maneuvering flight at wing angles greater than 40°. Reference 6 goes on to state that the limitation on longitudinal maneuvering in the low-speed powered-lift regime was not structural or aerodynamic, but rather a disorientation due to the high pitch and yaw rates encountered. The change in pitch rate with stick displacement was considered to be a more meaningful criterion for longitudinal maneuvering than the stick force per g normally associated with aerodynamic flight. The nose-up pitching velocity increased linearly with aft longitudinal stick displacement for the first inch of control travel (fig. 41). For step inputs larger than 1 inch, the gradient appeared to become nonlinear, with a greater increase in pitch rate resulting for a given control input; however, these characteristics were satisfactory (pilot rating of 2).

The X-22 longitudinal control characteristics in maneuvering flight (ref. 19) were investigated during powered-lift and aerodynamic flight. As shown in figures 42 and 43, the variation of longitudinal control position gradient with normal acceleration was positive but weak at zero duct angle (aerodynamic flight) and became more positive with increasing duct angle (powered-lift flight). As duct angle was increased from 0° to 60°, less normal acceleration was available and the cue most obvious to the pilot was pitch rate. This variable replaced normal acceleration for graphical presentation of test results. The gradient of longitudinal stick position versus steady-state pitch rate was linear and positive but weak (0.1 to 0.3 in./deg/sec) as shown in figure 44. Aft pitch control step inputs, with duct angles greater than 30°, resulted in considerable overshoot in pitch rate; in some cases, the peak pitch rate was twice the final steady-state value. Reference 19 goes on to state that the longitudinal maneuvering stability characteristics of the X-22, while not satisfactory for a transport aircraft, are acceptable for the flight research role; a pilot rating of 4 was assigned.

2.8 DYNAMIC STABILITY

The longitudinal dynamic stability criteria of part 1 require that the response of the aircraft should not be divergent (i.e., all roots of the longitudinal characteristics equation should be stable). In addition, the damping ratio of the second-order pair of roots that primarily determine the short-term response of angle of attack and pitch attitude following an abrupt pitch input should be at least 0.3 for the most critical undamped natural frequency.

These criteria assure that the response of the aircraft to a control input will not be unstable. That is, all roots should be at least neutrally damped. In addition, the short period response should have a damping ration of at least 0.3, which has been established from conventional flight. It was realized that some V/STOL aircraft do not have a clearly defined short period and phugoid response. It is the responsibility of the user, therefore, to determine if the oscillation in question is a short period response and if the 0.3 damping ratio criterion should be applied. The criterion is aimed at any oscillation that occurs at constant airspeed with a change in angle of attack. It is anticipated that the lowest frequency at which this will occur will be somewhere between 1 and 2 rad/sec with a period of 3 to 6 seconds.

The dynamic stability criteria of section 2.8 also require that the frequency and damping ratio of any oscillation superimposed on the normal control modes (i.e., by gusty air or poor SAS characteristics) should meet at least the values shown in figure 45 for VTOL aircraft in hover and V/STOL aircraft at the approach reference speed. Any sustained residual oscillation should not degrade the pilot's ability to perform the required task.

These dynamic longitudinal stability criteria are similar to those presented in the original AGARD 408 report but have been revised as a result of an investigation conducted by the Canadian National Aeronautics Establishment. The flight trials were accomplished using a variable stability helicopter that utilizes the "model-controlled" method of simulation. The model was programmed for a configuration with pleasant handling qualities plus an oscillatory response of adjustable natural frequency and damping ratio. The oscillation was superimposed on the basic control mode (which was first order) and was excited in a realistic manner as the pilot performed a hovering and low-speed maneuvering flight task. The pilot opinion data obtained are presented in figure 46 for various combinations of damping ratio (ζ) and natural period (τ_n). Boundaries for normal operation and for flight following a single failure (corresponding to the 3-1/2 and 6-1/2 pilot rating contours, respectively) have been faired through the data, yielding the curve of figure 46. As can be seen in the figure, the pilots were willing to accept lower levels of damping ratio for longitudinal oscillations than those previously specified in AGARD 408. The damping ratios required within the range of T = 2 to 20 seconds were as much as 0.2 less than the AGARD 408 "Normal Flight Limit" and the difference became greater at longer periods. These curves of damping ratio versus natural period were replotted on the S-plane form (fig. 45). Further confirmation for reduced values of damping at the lower frequencies is given in the results of the variable stability X-22A (ref. 21).

2.9 LONGITUDINAL CONTROL CHARACTERISTICS IN TAKEOFF

The intent of the first criterion for longitudinal control in takeoff is to bring out the need for the pilot to be able to adjust aircraft attitude sufficiently in advance of liftoff to take advantage of the performance potential of the aircraft. Pilots' comments on takeoff procedures for aircraft with high thrust/weight ratios point out the difficulty of making precise attitude adjustments when the total takeoff time duration is extremely short and monitoring airspeed becomes more difficult. For these reasons, the criterion was presented in terms of a time margin of 2.0 seconds for adjustment of attitude rather than a percentage of liftoff speed. Admittedly, the 2.0-second time value is subjective and is not supported by systematic tests, yet it is the commonly accepted time duration for other ground handling operation such as selection of reversed thrust, engine failure recognition, etc.

The force limitations indicated in the criterion of 25-lb pull and 15-lb push are to assure that forces should be moderate to maintain satisfactory precision of control and reduce pilot fatigue.

The requirement to prevent movement on the ground is self evident and certainly the aircraft should not skid across the ground during engine run-up. The background for this originated from early experience with the SC-1 (which initially had a castering gear) and from experience with the X-14A. Adjustment of aircraft attitude to maintain a given spot on the ground will vary with each V/STOL concept and depend on the magnitude of the ground reaction moment resulting from the landing gear geometry.

2.10 LONGITUDINAL CONTROL CHARACTERISTICS IN SIDESLIP

The criterion for longitudinal control in sideslip is intended to provide limits on cross-coupling effects during the critical phase of landing in cross winds. In general, these requirements that stem from operation of all types of V/STOL aircraft are self evident.

2.11 LONGITUDINAL CONTROL CHARACTERISTICS IN LANDING

The criterion for pitch control in landing provides for adequate control for adjusting aircraft attitude as desired for touchdown. The touchdown attitude adjustment may be more or less severe, depending on the specific pitching moments associated with ground effect for each configuration. Because these effects may vary between various VTOL configurations, and since no systematic tests on this subject have been conducted, no quantitative values for margins are specified.

Section 3.0

LATERAL-DIRECTIONAL STABILITY AND CONTROL

3.1 GENERAL

Flight operation of a wide variety of V/STOL aircraft has shown that pilots have been more critical of lateral-directional behavior and control about the roll axis than about any other axis. This is due partly because, for V/STOL operation, lateral positioning must be quick and precise and partly because of the more pronounced effect of cross winds during landing. Precise roll control is essential during landing approach because even small bank angles result in relatively large heading changes and sideward velocities at low speeds. Further, cross-coupling effects are more prevalent and many separate items can interact to determine the overall roll response apparent to the pilot. These factors make it more difficult to accurately specify values for handling qualities parameters because, in the usual case, the effects of the interactions are studied by varying one parameter while keeping all others at a satisfactory or optimum level. For example, systematic piloted simulator studies of roll control power are generally made with good values of control force gradient, friction, breakout, etc., optimum values of sensitivity, without adverse yaw or other degrading cross-coupling characteristics, low control system time lags, and good spiral and Dutch roll behavior. In a flight evaluation of a specific aircraft in landing approach, however, the pilot, in rating roll control power, is confronted with nonoptimum force characteristics, nonlinear control sensitivity, and poor turn coordination characteristics, but good Dutch roll damping. His comment may be that the aircraft has good control power because large angular accelerations are available, but overall heading control is unsatisfactory because large heading changes cannot be made with the desired accuracy.

In spite of the large amount of data accumulated from many specialized studies, additional work is needed to quantify the effects of the interrelated factors.

3.2 ROLL CONTROL POWER

The criteria of section 3.2 (ref. 3) require that from trimmed conditions specified for each aircraft, the roll control should be sufficient to achieve the response values shown in table 3.1. Aircraft whose missions require extensive maneuvering should be capable of at least the larger values indicated, while those for which maneuvering is only incidental to the mission and those for which direct side force control can also be used should be capable of at least the lower value noted.

The maximum cockpit control force to achieve the total values shown in table 3.1 should not exceed 10 lb for hover and 20 lb for STOL.

Table 3.1
Roll Control Power Characteristics

Parameter to be measured	Control power required for	Type of control system	Minimum levels for satisfactory operation	
			Hover	STOL
Roll angular acceleration, rad/sec^2	Maneuvering	Attitude command	0.2-0.4	0.1-0.6
		Rate	0.2-0.4	
		Acceleration	0.3-0.6	
Bank angle after 1 sec, deg	Maneuvering	Attitude command		
		Rate	2-4	2-4
		Acceleration	2-4	2-4
Roll control deflection at zero rolling velocity, in.	Trim	All	Sufficient control in EXCESS of MANEUVERING REQUIREMENTS to trim over designated speed and cg range and for most critical engine failure	
Time to recover to initial attitude or control deflection, sec	Upset (due to gusts, recirculation, ground effect, etc.)	All	Sufficient control in EXCESS of MANEUVERING and TRIM REQUIREMENTS to balance moment due to a specific gust; e.g., 30-ft/sec gust	
			Building up in 1 sec	Building over a 100-ft distance
Roll angular acceleration, rad/sec^2	Typical range of values used by V/STOL aircraft for maneuvering, trim, and upset	Attitude command	0.4-1.5	0.2-2.0
		Rate	0.8-2.0	0.3-2.5
		Acceleration	0.8-2.0	---

The pilot desires specific values of roll control for maneuvering, for trimming in sideward flight, and for controlling upsets due to turbulence or self-generated disturbances. Control power requirements depend on many factors: (1) the mission to be performed, (2) the susceptibility of a particular configuration to unsymmetric momemts resulting from aerodynamic or thrust-induced cross flow as well as turbulence and ground-induced disturbances, (3) aircraft size since, in general, the pilot tends to maneuver large aircraft less briskly and because of their higher inertias, tend to be disturbed less by turbulence, (4) the type of control system used, and (5) the amount of angular rate damping available.

Because of the foregoing considerations, the criteria for roll control power were broken down in the form shown in table 3.1. The chief purpose in separating the requirements is to force the user to examine how each one affects his particular aircraft design or flight evaluation. Different values of roll acceleration are given to take into account the type of control system used and the type of operation, that is, VTOL or STOL.

3.2.1 Control Required for Maneuvering. Note in table 3.1 that a range of values is given for minimum maneuvering control requirements to reflect mission requirements. Aircraft whose missions require minor maneuvering should be capable of at least the larger values indicated, while those for which maneuvering is only incidental to the mission and those for which direct side-force control can also be used should be capable of at least the lower value noted. The validity of the values shown is certainly open to question because they ultimately must come from real operational experience with different classes of V/STOL aircraft. Until these results are available, we can only speculate on the basis of limited data obtained from the various V/STOL aircraft, some of which have attempted to simulate operational-type maneuvers. Further, there is the problem of sorting, from data obtained during these maneuvers, the amount of control used uniquely for maneuvering and that which was used concurrently to correct for trim and upset due to gusts, turbulence, recirculation, etc. Perhaps the best answers can be derived from examining records of aircraft for which trim changes, by virtue of their engine and aerodynamic layout, are minimum. Further, if these aircraft use an attitude command type of control system, external disturbance effects are minimized. Results from two such aircraft, the VJ-101 and the DO-31 (ref. 22), show that satisfactory operation in hover was obtained with values of 0.2 and 0.4 rad/sec^2, respectively (fig. 47). The data presented in figure 48 substantiate the fact that the values listed in table 3.1 for maneuvering are adequate. It must be emphasized again that these are minimum values based on calculated responses. No attempt has been made to select optimum values for maneuvering because of the lack of operational experience; however, tests conducted in reference 13 suggest that values approximately twice those listed in table 3.1 may be desired for maneuvering. Confirmation of the lower value of roll angular acceleration for STOL operation has been obtained from "flights" in a piloted motion simulator (ref. 23). A slightly higher value (0.6 rad/sec^2) was selected for the upper end of STOL operation to reflect the need for more agile maneuvering into confined areas.

3.2.2 Control Required for Trim. Various amounts of roll control moment are needed for trim to maintain desired velocities in sideward (or sideslipping) flight. The amount of trim differs for each VTOL concept because of the difference in magnitude of rolling moment introduced from both aerodynamic and engine-induced flow sources. For aircraft with inherently large rolling moments induced by side velocity, ample control moment is needed to avoid the development of excessively large bank angles that may occur very abruptly with a loss in altitude when the aircraft is turned sideward from a headwind approach. Some types of V/STOL aircraft require that any asymmetric rolling moments associated with powerplant failure be trimmed out. Further, amounts of trim required depend on the cross-wind magnitudes specified for a particular mission and VTOL concept.

3.2.3 Control Required for Gusts. The amount of control power available to counteract upset due to gusty air or self-induced flow effects in ground proximity (which are also configuration dependent) directly affects the precision of the approach and touchdown. In vertical takeoffs and landings, the pilot may need to adjust attitude rapidly to avoid excessive side drift. Bank-angle excursions are undesirable in STOL approaches because of the tendency to induce large heading errors. In these cases, the pilot is interested primarily in returning to the desired bank angle in a given time. In addition, the type of control system used has a pronounced effect on control power requirements for upset. More sophisticated control systems such as attitude command, automatically reduce or eliminate the need for the pilot to correct for the upset. Because corrections can be sensed and made more quickly by the SAS, large bank excursions do not develop, with a resultant savings in control power.

The amount of control needed to handle upset due to gusts depends on two factors: (1) the sensitivity of a particular V/STOL concept to upset as influenced by wing loading (L/α), induced flow effects, inherent side-force characteristics, etc., and (2) the magnitude and character of the disturbing influence. It is in this latter area that the proposed criteria are weak. Although improvements have been made in gust measurement techniques, data analysis, and prediction effects, a well-defined gust model suitable for hover and STOL operation still remains to be defined. The criteria for upset used in table 3.1 attempt to establish a base from which firmer values can grow. It was considered necessary to specify a discrete gust effect rather than the usual rms (random noise type) to provide meaningful results for control power assessments.

The range of values for total control power given in the lower part of table 3.1 reflects the requirements for all the factors previously discussed and also the speculative nature of the criteria. The intent is to show typical ranges of total roll control power used by various aircraft as a guide for design purposes and not to represent firm numbers that must be met.

In the following paragraphs, an examination of flight-test data (a tabulation of roll control power data of various V/STOL aircraft is given in figures 49 and 50) has been made to show how various items interact to produce a given overall impression of roll response to the pilot. Shown in figure 51 are results of STOL aircraft tests (taken from ref. 4) obtained during approach and takeoff. The results are presented in terms of maximum angular acceleration obtainable as measured by the conventional roll reversal technique. For convenience, the data are presented as a function of gross weight which was used as a sizing formula $(W + 1000)^{1/3}$ in AGARD Report 408A (ref. 2). Included in the figure are the pilot's ratings of the overall roll response for each aircraft. It should be recognized, however, that angular acceleration is only a convenient parameter to use as a yardstick and it relates only indirectly to the pilot's impression of controllability. Further, when weight is used as a parameter, it only approximates the effects of size and, as noted previously, reflects maneuverability requirements and sensitivity to turbulence.

Note first that a large acceleration value does not necessarily indicate satisfactory pilot impression of roll response. For example, the VZ-3 aircraft has over three times the roll acceleration capability of the majority of the aircraft and yet has only a pilot rating of 4. The ability to maintain a desired bank angle while maneuvering in turbulence has been the most critical requirement for roll control of most of these STOL aircraft at takeoff and landing speeds. As an example, in tests of the BR 941, less than 40 percent of the available control was used during extensive maneuvering. Remember that this aircraft requires little lateral trim for cross-wind operation and the propellers are interconnected to remove any engine-out asymmetry trim requirements. The 941 is perhaps the best documented of all the STOL aircraft, having been flight tested with several lateral control modifications and extensively investigated in piloted motion simulators. Tests with this aircraft in IFR operation and moderate turbulence (ref. 24) indicated that satisfactory roll control (PR= 3) was obtained with a control power of 0.4 rad/sec^2 under these more adverse conditions. Note that for a heavier aircraft, the NC-130B, poorer ratings are evident for this same control power value which was based again on IFR operation in gusty air (ref. 25). The reason for degrading the overall roll controllability was due in part to low control sensitivity and the fact that at 70 knots almost

full roll control was required to trim for an inoperative engine, therefore, leaving too small a margin for maneuvering. The heaviest (and largest) aircraft tested was the 367-80 (707 jet transport) modified to incorporate a high lift BLC flap system (ref. 26). With the combined aileron spoiler system, the roll acceleration produced by large control deflections was so large for that size aircraft that the pilot was concerned about possible structural damage. In the initial tests with this aircraft, the ailerons were equipped with an aerodynamic tab control that was rated unsatisfactory (PR = 4-1/2) because of high forces and nonlinear response characteristics. Changing to a hydraulic powered control system with essentially the same rolling moment capabilities improved the pilot rating. These data also show that an improved pilot rating resulted when a higher approach speed was used, even though less acceleration was available. In this case, the cross-coupling effects ($C_{N_{\delta_a}}$, C_{l_β}, C_{n_p}) were greatly reduced at the lower C_L associated with the higher approach speed. A further example of interrelated effects is brought out by results obtained on the BLC equipped NC-134A aircraft (ref. 27). Even though very large lateral acceleration was available with the combined spoiler and aileron combination, precise use of this capability was difficult due to the nonlinear response. At 30° wheel position, the region most frequently used in controlling the aircraft, the rapid increase in response and the large increase in force when the spoilers were engaged combined to produce unsatisfactory characteristics that masked the control power usefulness of this aircraft.

The same parameters for VTOL aircraft in hover are presented in figure 52. Note that a wide range of values exists for the various aircraft and is generally well above the former AGARD sizing formula $(W \div 1000)^{1/3}$, which was really meant to be a minimum maneuvering requirement. Because of the lack of clarity in this respect, it was conveniently used in many paper designs (and for a few aircraft) as the total control power needed. It is obvious that a sizing rule is difficult to establish from these data for the reasons discussed in the following paragraph.

One of the first points to note is that the X-14A which has 1/4 the weight of the P-1127, can get by with less control power chiefly because the P-1127 requires a major portion of its available roll moment to trim for sideward flight. In fact, for the Harrier, sideslip is restricted in forward flight by a warning device on the rudder pedals (ref. 28). Further, even more roll control power would have been required for this aircraft had it not been for the fact that the control sensitivity and the mechanical characteristics of the control system were optimized for low-speed flight. Other aircraft that also require a large percentage of available control power to offset rolling moments associated either with sideward flight in hover or sideslip in forward flight are the XV-5B, SC-1, Balzac, and Mirage III-V. In fact, this particular trim requirement had been seriously overlooked in operational testing and, as a result, all the aforementioned jet aircraft (except the X-14A) have been damaged in accidents due to this trim problem, some seriously enough to be fatal. The major rolling moment contribution has come from induced flow effects associated with inboard locations of the jet engines. Notable jet lift VTOL aircraft that are exceptions to the sideslip trim problems are the VJ-101 and the DO-31, both of which have the jet engines located at the wing tips. This is reflected in the control power usage for the VJ-101 (ref. 14, which shows that only 0.25 rad/sec^2 was needed for roll control in typical takeoff and landing maneuvers. Similarly, with the DO-31, roll control power requirements for IFR approaches in gusty air showed that 0.4 rad/sec^2 was adequate. Both aircraft have larger roll control power available because of engine-out trim requirements.

3.3 TRANSLATIONAL CONTROL

The criteria of section 3.3 (ref. 3) state that for aircraft that use direct side force as a primary lateral positioning device, starting from trimmed conditions in hover for the designated wind conditions, the direct side-force control should be sufficiently powerful to obtain lateral acceleration values between 0.03 to 0.12 g in wings-level sideward flight.

The roll control power requirements for large VTOL aircraft, particularly those with high moments of inertia and large sensitivity to lateral gusts, may be unduly penalized if roll attitude is the only means of providing maneuvering and trim functions. The capability for wings-level translation, which would eliminate the need for rolling the aircraft to achieve a sideward thrust component, has been investigated using the X-14A (4000 lb) jet lift aircraft in flight and on a 6-degree-of-freedom motion piloted simulator (ref. 29). The sideward thrust component was obtained by a vane in the jet engine exhaust. Of interest in these studies was the determination of a satisfactory cockpit control for adjusting sideways translation, the magnitude of side acceleration desired, and the effect of side acceleration on reducing the amount of roll acceleration still required for attitude control.

The simulator study that preceded the flight tests considered three methods of controlling the vane that generated the side force: vane deflection was programmed proportional to control stick deflection; vane deflection was programmed proportional to roll attitude; and vane deflection was commanded by a separate thumb controller located on top of the control stick. With the first method, no combination of side-force control parameters and basic roll control system parameters could be found that did not introduce control phasing problems. The second method has the effect of increasing the sensitivity of side acceleration due to bank angle as seen in the expression:

$$Ay = K(g \sin \phi)$$

where K is normally equal to unity. Gains up to $K = 1.5$ were considered helpful, but at higher gains the tendency of pilot to overcontrol began to degrade the system excessively. For the third method, two types of thumb controller action - on/off and proportional - were studied. The proportional control provides a vane angle proportional to switch position and a variable rate of thrust vectoring proportional to the rate of thumb controller displacement. The proportional thumb controller was preferred over the on/off switch because of the pilot's desire to modulate side accelerations for precise control. The proportional system also permitted slightly higher control authority to be utilized comfortably because of the more gentle onset of side acceleration.

The results showed that the pilots preferred a thumb controller on top of the stick designed to give sideward acceleration proportional to control deflection (see section 1.8). This allowed precise modulation

of side acceleration in contrast to that obtained with an off/on-type controller. The minimum magnitude of side acceleration desired by the pilot is approximately 0.1 g (fig. 53). The effect on reducing roll acceleration requirements is shown in figure 54. Conclusions from this study were the following: (1) for a simple hovering task, the direct side-force control was preferable to using roll when only low values of roll control power were available, (2) for a complex maneuvering task involving forward motion of the aircraft, the side-force controller introduced skidding turns that were unnatural for the pilot. Further, information is needed for side-force controls on attitude-stabilized aircraft before generalizations to other aircraft can be made.

3.4 ROLL CONTROL SENSITIVITY

The criteria of section 3.4 require that it should be possible to achieve at least the values shown in table 3.2 following an abrupt 1-in. step of the cockpit roll control. These criteria apply starting from nonaccelerated flight for the designated wind conditions in hover and at selected reference approach speeds in STOL operation. In addition, the linearity of roll response with cockpit roll control deflection should be in the direction noted in table 3.2.

Table 3.2
Roll Control Sensitivity Characteristics

Item	Parameter to be measured	Type of control system	Minimum levels for satisfactory operation	
			Hover	STOL
Sensitivity	Attitude change per unit control deflection, deg/in.	Attitude command	3-3	
	Roll angular acceleration per unit control deflection, rad/sec^2/in.	Rate	0.15-0.30	0.05-0.25
	Roll angular acceleration per unit control deflection, rad/sec^2/in.	Acceleration	0.2-0.8	
Linearity	Variation of roll angular acceleration with control deflection	All	Constant or should not abruptly increase nor change sign	

Many studies have been conducted over the years to establish control sensitivity requirements for hover and STOL operation. Optimum roll control sensitivity depends on several factors: the amount of control power available, the amount of angular rate damping, the type of control system used, the sensitivity of a specific aircraft to turbulence and self-generated (ground effect) disturbances, the magnitude of rolling moment associated with side velocity and sideslip (in forward flight), and the magnitude of side force with side velocity.

The values of control sensitivity are listed in table 3.2 to cover a range having possible satisfactory characteristics. Although only the different types of control systems are identified as influencing the values required, as previously noted several other factors must be considered. Only minimum values of sensitivity have been designated for the various types of controls systems considered. An upper limit on sensitivity, usually dictated by PIO tendencies, has not been designated. Because many factors such as low force gradients, control lag, friction, etc., combine to produce PIO tendencies at high values of sensitivity, it is not possible to generalize on these points. The roll control sensitivity values for various VTOL and STOL aircraft are presented in figures 49 and 50.

One of the earliest studies (ref. 30) showed the dependence of control sensitivity on the amount of angular rate damping. These results (fig. 55) show that satisfactory sensitivity values of the order of 0.2 to 0.4 rad/sec^2/in. occurred at the lower values of damping. Lower values of sensitivity at a given value of damping result in sluggish vehicle response while larger values can result in overcontrolling tendencies. These results were obtained using a small single-rotor helicopter modified for variable stability operation. This study considered only rate-damped control systems and inherently had limitations in its capability to represent all types of aerodynamic and turbulence sensitivity problems typical of other nonhelicopter-type V/STOL aircraft.

Flight studies using the X-14A, a small twin-jet VTOL aircraft, modified to incorporate variation in control power, sensitivity, and rate damping have been reported in references 31 and 32. These tests were flown under calm air conditions and represent sensitivity requirements for maneuvering, trim, and upset due to recirculation in ground effect. The values for satisfactory lateral control sensitivity obtained from the hovering and side-step maneuvers fall in the range from 0.4 to 0.8 rad/sec^2/in., depending on the angular rate damping used (figs. 56 and 57). Another study with a jet lift VTOL aircraft, the S.C. 1, equipped with variable stability and control features (ref. 33) showed sensitivity values about the same as the X-14A (control sensitivities of 0.4 rad/sec^2/in.). Pilots' comments indicated that increased sensitivity would have been desirable at the midtravel control position. Even higher values of control sensitivity were judged desirable in hover tests of the jet lift P-1127 (Kestral) aircraft (ref. 7). This aircraft was tested without augmented rate damping and the values of roll control sensitivity of 0.65 rad/sec^2/in. reflect the need for brisk response to be able to quickly correct for lateral disturbances due to gusts or recirculation in ground effect.

Piloted simulator 6-degree-of-freedom motion results (ref. 34) for the hover task showed the effects of different types of control systems on control sensitivity and control power requirements. These results show that satisfactory ratings were obtained for control sensitivity values in the range of 0.4 to 0.8 rad/sec^2/in. regardless of the type of control system used (figs. 58, 59, and 60). These tests were conducted with random disturbance inputs only. Pilots' comments indicated the desirability for increased control sensitivity and damping with increased intensity of disturbances. With the attitude command system, the disturbances were not as apparent to the pilot; however, increased control sensitivity was required to overcome the "stiffness" of aircraft response characteristic of operation at high control system frequencies. The upper values of satisfactory control sensitivity for the attitude system at high values of frequency was 3.5 deg/in. Higher sensitivity values are needed for lower frequency control systems and when realistic environmental conditions of turbulence and counteracting sudden trim changes such as those caused by engine failure are considered. This has been verified by unpublished flight-test results on the VFW 1262 hover rig and by operation of the VJ-101 and DO-31 aircraft (ref. 22).

An examination of control sensitivity values for STOL aircraft taken from reference 4 (fig. 50) showed that in no case were adverse comments raised about too high sensitivity values around neutral control position; however, there were several adverse comments about too low sensitivity. These data show that sensitivities as low as 0.05 rad/sec^2/in. were considered satisfactory for aircraft that were not required to be maneuvered extensively during landing. The highest value of sensitivity available on any of the STOL aircraft tested (ref. 4) was 0.18 rad/sec^2/in. for the VZ-3RY and CV-48 aircraft.

Other studies (ref. 35) have shown that for lighter wing loading aircraft typical of General Aviation types, much higher sensitivities (of the order of 1.7 rad/sec^2/in.) were needed. A factor causing this very large value to be required was the lateral gust response associated with the large dihedral effect (L_β = -16 (rad/sec^2/rad). In view of the possible need to provide control sensitivities for the lighter more gust sensitive aircraft, the upper value of 0.25 rad/sec^2/in. is given. Note again that the values shown represent minimum values.

3.4.1 Linearity of Roll Response. As stated in table 3.2 of reference 3, the variation of roll angular acceleration with cockpit control deflection should be constant and not abruptly increase or change sign. The background for this requirement came from tests of aircraft (YC-134A, C-8A, and BLC equipped 367-80 (ref. 4)) wherein a marked increase in response occurred during a midportion of control travel. This abrupt increase in roll control sensitivity, which was due to activation of spoiler surfaces for roll control, caused the pilot to overshoot or undershoot his intended roll angle. In contrast to this type of nonlinearity, the Breguet 941 aircraft (ref. 4) had a nonlinear response that gave higher control sensitivity at small roll control deflections than at large deflections. This latter type of nonlinearity was considered satisfactory. A further verification of this condition was made in comparing the roll response of the C-8A and BR 941 aircraft (ref. 4), which both had the same total roll control power; the C-8A was rated unsatisfactory partly because of the nonlinear roll response. Subsequent modifications to the lateral control of the C-8 aircraft resulted in a nonlinear control response similar to that of the BR 941, and these characteristics were considered satisfactory.

A further qualification on linearity of response to avoid change of sign have been given to avoid those cases where the initial roll response may occur in the wrong direction.

3.5 CROSS-COUPLING

The criteria of section 3.5 (ref. 3) state that at trimmed conditions at the selected reference approach speeds in STOL operation, the magnitude of cross-coupling about all axes following abrupt-roll control inputs (yaw control free) up to that required to meet the roll control criteria should have the characteristics noted in table 3.3.

Table 3.3
Cross-Coupling Characteristics in STOL Operation
(Yaw Control Free)

Parameter to be measured	Level for satisfactory operation
Maximum sideslip-to-bank-angle ratio $(\Delta\beta/\Delta\phi)_{max}$ or maximum sideslip angle	Not greater than approximately 0.3 to 0.5 and not to exceed 20° sideslip angle
Pitch angle change $\Delta\theta$	Not objectionable
Normal acceleration change Δg	0.1 g

Because of reduced directional stability and damping at low speeds, moments generated by roll control inputs tend to result in larger sideslip angles than in conventional flight. These sideslip angles that result from the yawing moment due to (1) roll control deflection and (2) due to roll rate are large at high lift coefficients; consequently, their influence is greater for STOL aircraft and they increase the requirements for turn coordination to reduce sideslip. The turn entry coordination problem is discussed in detail in reference 36 and illustrated in figures 61 and 62, which shows time histories of roll maneuvers performed with the NC-130B aircraft at 70 knots. These results show that although the desired bank angle was obtained in 2-1/2 sec, 7 seconds elapsed before the heading changed in the correct direction. When the pilot attempts to coordinate the turn, he must supply different amounts and phasing of the rudder to account for the effects of adverse yaw, yaw due to roll rate, and yaw rate damping which occur at different times during the turn.

The cross-coupling that occurs when using roll control has been expressed as a ratio of maximum sideslip angle to bank angle $(\Delta\beta/\Delta\phi)$ in table 3.3. This relationship was selected because the pilot is willing to accept larger absolute values of sideslip when large bank angles are being used.

The cross-coupling parameter, $\Delta\beta/\Delta\phi$, is measured during an abrupt bank-angle change with rudder fixed. Correlation of $\Delta\beta/\Delta\phi$ with pilot rating of turn coordination is given in figure 63 for various aircraft and for a range of lateral-directional characteristics studied on the simulator. These data indicate that values of $\Delta\beta/\Delta\phi$ less than approximately 0.3 were rated satisfactory (pilot rating of 3-1/2). The values shown for the various STOL aircraft point out the need for augmentation when operating at STOL approach speeds. Improvements are noted for the NC-130B and 367-80 aircraft by the addition of positive N_p and N_β augmentation (ref. 37).

The lag in heading change measured from initiation of control input (previously pointed out in the discussion of the NC-130B aircraft) has been recognized as a major part of the turn coordination problem. It has not been possible, however, to develop a criterion based on heading lag alone as there is a significant interaction between roll and heading control, depending on the roll-mode time constant (see ref. 38). The simulator tests indicated that when good roll damping existed, a larger heading lag was tolerable and vice versa. These results suggested a relationship between heading lag and roll-mode time constant (fig. 64).

3.6 ROLL ANGULAR DAMPING

The criteria of section 3.6 (ref. 3) state that for the flight condition specified in hover and STOL operation, the aircraft should possess the roll angular velocity damping characteristics of at least the lower values shown in table 3.4.

Table 3.4
Roll Angular Velocity Damping

Parameter to be measured	Type of control system	Minimum levels for satisfactory operation	
		Hover	STOL
Angular velocity damping, 1/sec	Attitude command	-1.5 to -4	-0.5 to -3
Damping ratio		<15 percent overshoot	
Angular velocity damping, 1/sec	Rate	-2 to -4	-0.5 to -3
Number of control reversals required to stabilize		Not more than 1	Not more than 2

The roll angular damping criteria shown in table 3.4 indicate only slightly different values, depending on the type of control system and the mode of flight operation. A range of values is presented to account for the different amounts of maneuvering and the sensitivity to upsets noted in previous discussions.

Roll angular damping expressed as the first-order roll time constant is one of the more significant handling qualities parameters because it defines the overall roll response of the aircraft and the sensitivity to asymmetric vertical or rolling gusts. Many experiments have been conducted to determine satisfactory levels for hovering and forward flight.

Lack of adequate roll rate damping is most noticeable in precision control tasks where the pilot must continually correct for divergencies in roll attitude. Although a few VTOL aircraft have been successfully flown with only inherent roll damping (most notably the P1127), most VTOL aircraft have used augmentation systems to increase damping and thereby reduce pilot workload in demanding tasks such as IFR approaches or in night flying. Loss of artificial roll damping is more serious for attitude command systems (see ref. 34) because of the difficulty in damping the residual oscillations inherent in the "springiness" of the attitude system. For this reason, a limit on the amount of overshoot is listed for these artificially stiffened control systems.

Roll damping in forward flight at STOL operating speeds is usually adequate because of the effects of aerodynamic lift supplied by the wing. Values of roll time constant from simulator studies (ref. 39) and for several STOL aircraft (ref. 4) are compared with pilot rating in figure 65. These data indicate that an upper time constant limit of 2.0 sec provided satisfactory damping. Reference 40 suggested that the roll time constant for transport aircraft should not exceed 2.0 to 3.0 sec for satisfactory damping.

In establishing the lower values of damping, consideration was given to the aircraft's sensitivity to upsets. An unaugmented STOL aircraft with a small roll time constant has a high aerodynamic damping that depends on the value of lift-curve slope. Unfortunately, the high lift curve slopes characteristics of current STOL aircraft also increase the magnitude of roll disturbances due to gusts. A conflict in requirement thus exists and a lower limit of -0.5 sec has been selected.

In table 3.4, an "apparent" roll damping expressed as the number of control reversals required to stabilize has been included also as a qualitative criterion for the rate control system. As discussed in reference 4, the pilot prefers to be able to easily arrest and stabilize an established roll rate. In certain cases, a continuous roll oscillation sustained by the pilot's control activity can occur with a combination of aerodynamic and mechanical control characteristics. Sufficient data are not available to quantify the interactive effects of the parameters involved and, therefore, "apparent" roll damping has been listed in qualitative form.

3.7 ROLL CONTROL SYSTEM TIME LAGS

The criteria of section 3.7 (ref. 3), require that following an abrupt displacement of the cockpit roll control, the angular acceleration response should be in the commanded direction within 0.1 sec, and the time required to reach 63 percent of the peak angular acceleration should meet the values listed in table 3.5.

Table 3.5
Roll Control Lags

Parameter to be measured	Type of control system	Levels for satisfactory operation	
		Hover	STOL
Time from control input to 63 percent of peak angular acceleration, sec	Attitude command	<0.2	<0.3
	Rate	<0.2	<0.3
Time to 90 percent of demanded attitude change, sec	Attitude command	>1 and <2	

The effects of moment generator lag on roll control of VTOL aircraft in hover were investigated on a piloted 6-degree-of-freedom motion simulator (described in ref. 41). First- and second-order lags were introduced into acceleration, rate, and attitude control systems and the resulting responses were evaluated for a hovering and a maneuvering task. The first- and second-order lags that were simulated can be written in Laplace transform notation:

$$\mathrm{lag}(S) = \frac{1}{T_{63}S+1} \qquad \text{(first order)}$$

$$\mathrm{lag}(S) = \frac{\omega_{n_L}^2}{S^2 + 2\zeta_L\omega_{n_L}S + \omega_{n_L}^2} \qquad \text{(second order)}$$

where T_{63} is the time required to reach 63 percent of the command output, ζ_L is the damping ratio of the lag, and ω_{n_L} is the undamped natural frequency of the lag. To ease the comparison between first- and second-order lags, ζ_L and ω_{n_L} were arbitrarily related to T_{63}. The relationship between T_{63} and ω_{n_L} for the second-order lag is shown in figure 66 for two values of ζ_L. The basic output form for second-order lags can be altered by varying the damping ratio to obtain different amounts of initial response overshoot.

The effects of first-order control moment lag on acceleration, rate, and attitude control systems are shown in figure 67 and discussed in ref. 42. The effect of lag on the acceleration system was an immediate increased tendency to overshoot bank angle even during mild maneuvering. In this case, it was more difficult for the pilot to evaluate the effects of lag because of the general unsteadiness of hovering with an unstabilized system. With rate and attitude systems, the pilot was able to immediately detect the introduction of lag because the stabilized aircraft could be hovered more easily. The lag could reach approximately 0.2 sec before the pilot rated the system unsatisfactory.

Figure 67 indicates that the attitude system suffered more from the first-order lag than did the rate system. A possible explanation for this result may be obtained from an examination of the root locus diagram shown in figures 68 and 69. Below T_{63} = 0.15 sec, the rate system exhibits a damping ratio greater than ζ = 0.7. The attitude system damping ratio, however, is less than ζ = 0.7 for any value of lag. This correlates directly with pilot comments, which were that the effects of first-order time lag were to decrease the apparent damping and to increase the tendency towards PIO. Because increased lag appeared to the pilot as decreased system damping, several runs were made with the stabilization damping increased from ζ = 0.7 to ζ = 1.0 and to ζ = 1.5. Doubling the damping had very little effect on the system and changed the pilot rating from 4 to 3-1/2 to 4. The last data point (ζ = 1.5) was also tested with increased control sensitivity but no significant improvement in the pilot rating was found.

The effects of second-order lag on an attitude and a rate system are shown in figure 70. For this plot, the damping ratio of the lag was maintained constant and, therefore, percent overshoot was constant at 4.6 percent while the natural frequency, ω_L, was changed to produce a range of values for T_{63}. As can be seen from the data, the rate and attitude systems were very similarily affected by the second-order lag. It is believed that this can be attributed to the fact that both systems are of higher order and are oscillatory at any value of lag.

Figure 71 is a plot of the results when the time to 63 percent is held constant (T_{63} = 0.12 sec) and percent overshoot was varied for an attitude system. Overshoot does not appear to be a significant factor so long as it is less than about 40 percent. For practical systems having a response lag of a second-order form, overshoots of 40 percent or greater seem unlikely. As in the case of lag itself, overshoot also appeared to the pilot as a reduction in the system damping ratio which resulted in an increased requirement for pilot compensation to maintain precise control.

Figure 72 compares the effect of first- and second-order control moment lag on a rate-stabilized control system. With values of T_{63} greater than 0.1 sec, the rate system was rapidly degraded because of the

increasing oscillatory characteristics and reduced damping. This break in the pilot rating curve was delayed for the control system with first-order lag until the response of the control system exhibited a damping ratio of less than 0.7 (fig. 68). When lag was introduced into attitude-stabilized control systems, there was no apparent difference between the two control lag types when percent overshoot of the second-order lag was kept small (fig. 73).

All the preceding data indicate that the critical time factor for lags is approximately 0.15 sec. Since the simulator results would tend to be conservative because of very rigid maneuvering limits, values listed in table 3.5 were relaxed to 0.2 sec for VTOL aircraft and 0.3 for STOL aircraft. These higher values agree with the documentation data presented in section 2.5 (Pitch Control System Time Lags) that were generated by a flight simulation study using a variable stability helicopter.

3.8 PEAK ROLL CONTROL FORCES

The criteria of section 3.8 (ref. 3) state that the cockpit roll control forces required to obtain the designated roll response should not exceed 15 lb for hover and 20 lb for STOL operation.

These relatively low maximum values are compatible with the force gradients and control gearing criteria set forth in sections 1.3 and 1.9 and the need for one-hand operation to perform required bank-angle maneuvers in hover and STOL operation. Slightly higher maximum forces are allowed for STOL operation since there would be less of a requirement for "tight" control when the aircraft is not close to the ground. No systematic tests were available to establish firm values for STOL aircraft; however, measurements made on several STOL aircraft (summarized in fig. 4) indicate the desirability of limiting the maximum force to approximately 20 lb.

3.9 SPIRAL STABILITY

The criteria of section 3.9 (ref. 3) state that with the aircraft trimmed for wings-level, zero-yaw-rate flight for all normal flight conditions up to V_{con}, the spiral characteristics should be such that with the lateral control free and following an intentional small bank-angle input ($\phi = 10°$), no increase in bank-angle buildup is desired; however, in no case should the bank angle double in less than 20 sec. In addition, there should be no objectionable coupling between the conventional roll and spiral modes.

Good spiral stability characteristics are essential for ensuring low divergence rates to reduce pilot workload during critical approach tasks. Although systematic research results are available from flight tests on conventional aircraft (see, e.g., ref. 43), little information is available from flight tests during STOL operation. The results of a piloted simulator study showing the effects of spiral stability on pilot rating have been summarized in reference 4 and are presented in figure 74. These data, as well as pilot comments from evaluations of the various aircraft shown in the figure, reflect the desirability of having near neutral spiral stability. These data indicate also that satisfactory pilot ratings are obtained when the bank angle does not double in less than 20 sec ($1/T_2 = -0.05$).

The apparent spiral behavior noted by the pilot can be greatly influenced by the mechanical control systems characteristics and the lateral-directional trim change with speed. The inability to precisely trim has long been recognized as a major cause of poor apparent spiral behavior. As discussed in more detail in reference 4, STOL aircraft such as the C-8A, which had poor control wheel centering, could not be trimmed laterally, requiring that the pilot constantly search for the correct roll control position to reduce spiral divergences.

A further problem of coupling can occur between the conventional roll and spiral modes on V/STOL aircraft. The results of a simulator study (ref. 44) point out that when these two modes couple, the pilot sees a serious degradation in roll damping.

3.10 DIHEDRAL EFFECT

The criteria of section 3.10 (ref. 3) state that for operation at and above the reference approach speed and for sideslip conditions specified, the rolling moment variation with sideslip should be such that for conventional control systems, left roll control deflection and force are required in left sideslip and vice versa.

The extent to which lateral-directional control coupling interferes with precise control of roll response is largely influenced by the dihedral effect characteristics. As pointed out in the flight-test studies of reference 45, large values of L_β result in large roll disturbances when flying in turbulence. At low values of L_β, the pilots objected to predominant yawing or "snaking" motions in the aircraft's response to gusts. Results presented in reference 4 and shown in figure 75 indicate, for STOL aircraft, the desirability of near neutral dihedral effect, particularly when Np is low or negative (the usual case for most straight wing aircraft). When Np is optimized for turn coordination, the pilot is more tolerant to increased L_β. A satisfactory balance between low L_β values for minimizing gust sensitivity and the higher values needed to avoid spiral instability may be difficult to achieve because of the many interrelated factors.

For reasons mentioned in the foregoing discussion, the criteria of section 3.10 (ref. 3) do not attempt to apply quantitative values to dihedral effect, but merely serve to try to obtain precise control by requiring correctness of sign, linearity, and optimum forces.

3.11 YAW CONTROL CHARACTERISTICS - GENERAL

Yaw control criteria must consider the different modes of operation from that of pure heading control in hover; for maintaining trim sideslip angle in forward flight, for engine-out, and turn entries; for a specific sideslip angle in cross-wind landings and to decrab (align) the aircraft in touchdown. Although heading errors can usually accumulate with relatively less regard for safety in flight, control of the yaw axis is nevertheless important in reducing undesirable roll cross-coupling effects and the need to minimize

heading deviations in IFR operation. Over the years, many studies have been directed at defining the requirements for yaw control by means of ground and airborne simulators and from tests of specific aircraft.

3.12 YAW CONTROL POWER

The criteria of section 3.12 (ref. 3) state that from trimmed conditions in hover, at the selected reference approach speeds in STOL operation, and for the wind conditions specified, the yaw control power should be sufficient to achieve at least the values of aircraft response shown in table 3.6.

Table 3.6
Yaw Control Power Characteristics

Parameter to be measured	Control power required for	Minimum levels for satisfactory operation	
		Hover	STOL
Yaw angular acceleration, rad/sec^2	Maneuvering	0.1-0.5	0.15-0.25
Time for 15° heading change, sec	Maneuvering	1.0-2.5	<2.0
Steady-state sideslip angle, deg	Trim	---	$\beta = \sin^{-1}\left(\frac{V_{cross\ wind}}{V_{approach}}\right)$
Yaw control deflection at zero yawing velocity, in.	Trim	Sufficient control in EXCESS of MANEUVERING REQUIREMENT to trim over designated speed and cg range and for most critical engine failure	
Time to recover to initial heading, sec or control deflection, in.	Upset (due to gusts, recirculation, ground effect, etc.)	Sufficient control in EXCESS of MANEUVERING and TRIM REQUIREMENTS to balance moment due to a specific gust; e.g., 30-ft/sec gust	
		Building up in 1 sec	Building over a 100-ft distance
Yaw angular acceleration, rad/sec^2	Typical range of values used by V/STOL aircraft for maneuvering, trim, and upset	0.35-0.8	

As previously discussed for the pitch and roll axes, maneuvering, trim, and control of upsets are also requirements for sizing the amount of yaw control power needed for operation of V/STOL aircraft. The maneuvering requirements can be expressed in two forms: (1) angular acceleration for an aid to the aircraft designer and (2) time to change heading 15°, a condition of primary interest to the pilot. The need for adequate angular acceleration for STOL aircraft is brought out by the requirement to decrab (align the aircraft) for touchdown during cross-wind approaches made at zero sideslip (crabbed approach). For hovering operation, the acceleration values should cover a wide range to reflect mission requirements.

The values given in table 3.6 were obtained from a number of flight and simulati udies (figs. 76 and 77). An angular acceleration capability of 0.38 rad/sec^2 was available on the P-1127 (ref. 7); this was considered adequate for normal VTOL maneuvering under conditions of no wind. Very low values of yaw acceleration were used on the Balzac and Mirage III-V aircraft where only 0.11 and 0.18 rad/sec^2 were available. The acceptability of the lower values shows some correlation with the results of reference 23 where angular accelerations of 0.2 rad/sec^2 were found satisfactory for maneuvering when requirements for trim or turbulence were minimized.

For STOL aircraft, the data summarized in reference 4 show that values of yaw acceleration of 0.15 were rated satisfactory for the largest aircraft tested. A requirement for a 15° heading change in no greater than 2.2 sec was established by these results. In addition, the "approach" results of reference 23 support the 0.15 rad/sec^2 value previously noted.

For the trim requirements in cross-wind operation, the STOL approach steady-state sideslip angle has been expressed in terms of the ratio of cross-wind to approach speed. These values, which are influenced by the mission requirements and selected by the user, will directly influence the vertical tail design to avoid stalling. Most of the STOL aircraft tested (ref. 4) were limited to sideslip angles less than 20° (fig. 78). Presumably, greater sideslip angles and therefore larger cross-wind operation would be achieved by leading-edge protection of the vertical fin or by touching down in a crabbed condition. The other critical trim requirement is that imposed by a propulsion system failure. This yaw requirement can be calculated, of course, from the geometry of each specific aircraft and wind-tunnel data.

The amount of control required to deal with upset in yaw due to turbulence and recirculation is difficult to specify; however, this requirement deserves considerable attention based on pilots comments of the yaw behavior of several V/STOL aircraft in close proximity to the ground. Although all V/STOL aircraft are disturbed to some degree in yaw by turbulence and recirculation effects, the most notable example of extreme yaw upset is that encountered by the XC-142 in ground effect during STOL operation (ref. 46).

Aircraft equipped with attitude command (heading hold) control systems such as the DO-31 and VJ-101 (ref. 22) tend to minimize the disturbances apparent to the pilot but increase turn coordination problems (see ref. 47). The total values of yaw control power noted at the bottom of table 3.6 reflect the requirements for all the factors previously discussed and represent the range of values used by V/STOL aircraft in a wide variety of investigations.

3.13 YAW CONTROL SENSITIVITY

The criteria of section 3.13 (ref. 3) state that it should be possible to achieve at least the values of response shown in table 3.7 following an abrupt 1-in. input of the yaw control. These criteria should be met starting from trimmed, zero-yaw-rate flight in hover and at the selected reference approach speeds in STOL operation.

The variation of yawing acceleration with yaw control deflection should be in the direction noted in table 3.7.

Table 3.7
Yaw Control Sensitivity Characteristics

Item	Type of control system	Parameter to be measured	Minimum levels for satisfactory operation	
			Hover	STOL
Sensitivity	Rate	Yaw angular acceleration per unit control deflection, rad/sec^2/in.	0.08-0.2	0.05-0.10
	Acceleration	Yaw angular acceleration per unit control deflection, rad/sec^2/in.	0.05-0.2	
Linearity	All	Variation of yaw angular acceleration with control deflection	Constant or should not abruptly increase nor change sign	

The yaw control sensitivity criteria presented in table 3.7 were derived from pilots' evaluations of simulator and flight studies of various V/STOL aircraft. The lower values of sensitivity would be considered adequate only for aircraft that do not require extensive hovering operation and have little tendency for upset due to gusts or ground-induced turbulences. Aircraft for which very low values were adequate include the Balzac (ref. 48) and VZ-4 (ref. 49), although admittedly the pilot preferred higher sensitivity values.

One of the critical factors that determines the amount of yaw control sensitivity required is the amount of directional (weathercock) stability present, since this and angular rate damping reflect the aircraft sensitivity to gusts and ground-induced disturbances. A systematic study of yaw control sensitivity requirements as influenced by weathercock stability, N_v, and damping is contained in reference 50. These tests conducted under VFR conditions with an airborne flying simulator (Bell Helicopter) very clearly demonstrated the need for increased control sensitivity as either damping or N_v was increased (fig. 79). For the smallest value of N_v tested, the results indicated a minimum yaw control sensitivity of approximately 0.2 rad/sec^2/in. A later study with this equipment (ref. 51) compared the yaw control criteria for visual and instrument flight operation. These results (fig. 80) showed that slightly higher values of control sensitivity were needed for satisfactory pilot ratings for the IFR task, particularly at the lower value of N_v.

For several STOL aircraft (ref. 4), available yaw control sensitivities ranged in value from 0.06 to 0.16 rad/sec^2/in. (fig. 77). These STOL aircraft, in general, were rated satisfactory with lower sensitivity values than that required for VTOL aircraft. This would be expected in part because of lower directional stability and higher yaw rate damping and therefore less gust sensitivity. The lower directional stability results from the destabilizing effect of the propellers used on the majority of these aircraft. Another factor is the greater tolerance toward less precision of control required at the higher altitudes normally associated with STOL operation.

Linearity of directional response (table 3.7) is similar to that suggested for roll response. Traditionally, handling qualities requirements have specified linear response up to 15° sideslip, allowing nonlinear characteristics thereafter. In the area near neutral yaw control deflection, and when fine heading adjustments are needed, the requirement must be more stringent. Although systematic tests have not been conducted to determine quantitative values for yaw response linearity, operational experience with the CL-84 V/STOL aircraft (ref. 6) pointed out that increased pilot workload and loss of precision resulted when nonlinear yaw response characteristics prevailed in the normal operating range of yaw control deflection.

3.14 CONTROL SYSTEM TIME LAG

The 0.3 sec value of yaw control system lag presented in section 3.14 (ref. 3) was obtained by generalizing the results of the studies conducted for the pitch and roll axes. Although no systematic test results are available, it was reasoned that a slightly longer first-order time constant could be tolerated in the yaw axis because comparable "tightness" of control is not required. Further, the yaw

axis is not as critical from the standpoint of safety in flight, as evidenced by the general relaxation of the criteria for yaw control.

3.15 PEAK YAW CONTROL FORCES

The criteria for yaw control forces specify upper and lower limits of 15 to 50 lb for hover and 50 to 100 lb for STOL operation. These values are compatible with the force gradients contained in section 1.3 and also reflect the consensus of pilot evaluation of STOL aircraft (ref. 4).

3.16 YAW CROSS-COUPLING

The criteria of section 3.16 state (ref. 3) that from trimmed conditions in hover and at the selected reference approach speeds in STOL operation, the magnitude of cross-coupling about all axes following yaw control input should meet the criteria shown in table 3.8.

Table 3.8
Yaw Cross-Coupling Characteristics

Parameter to be measured	Levels for satisfactory operation	
	Hover	STOL
Apparent dihedral, rolling moment variation with yaw rate	Positive, but not requiring more than 50 percent of roll control to trim for $\delta_{r_{max}}$	
Response about pitch axis	Pitch-angle change less than 2°	Not objectionable; pitch-down with increase in sideslip
Response in vertical direction	Less than 0.05 g	Not noticeable

Of primary interest for V/STOL aircraft is the apparent dihedral effect generated by yawing velocity and resulting sideslip. This condition can be generated in hover operations by turning out of a headwind or by making wings-level (skidding) turns in air taxi operations. The primary intent of the criteria is to limit the positive dihedral effect rather than establish a replacement for raising the wing with the yaw control. In fact the magnitude of rolling moment generated by yaw control when used to change heading at low speeds in a skidding turn or to raise the wing in correcting for a roll upset due to turbulence can be quite serious for V/STOL aircraft, particularly the jet-lift type. The limitation on the amount of apparent dihedral effect expressed in terms of a 50-percent roll control margin is somewhat arbitrary and may require modification at some future date. Undoubtedly, some successfully demonstrated V/STOL aircraft do not meet this criterion even in hover, much less in forward flight where positive dihedral effect becomes more severe, particularly at high angles of attack. Nevertheless, the consequences of excessive yaw-roll cross-coupling have been overlooked in operational testing, and aircraft such as the Balzac, Mirage III-V, SC-1, P-1127, and XV-5A have been damaged in accidents caused by this yaw-roll coupling problem. Adding to the severity of the problem is the fact that the pilot may not react instinctively to apply the correct control input to reduce the roll excursion. For example, use of the roll control to reduce the bank-angle divergence resulting from the positive dihedral effect generated in a skidding turn may only aggravate the situation because of adverse yaw. When the correct yaw control input is made to reduce the sideslip angle, the problem is greatly minimized. A method to keep sideslip angle within prescribed limits is used on the Harrier Jet VTOL aircraft in which rudder pedal shakers are provided to serve as a warning when dangerous values of sideslip are being approached.

Although no yaw-roll coupling criteria were presented in table 3.8 for STOL operation, this was not intended to imply that they may not be needed; but that insufficient data are available for realistic values. Certainly, the tendency to make skidding turns is less for higher speed flight (due to positive directional stability). Further, for most STOL aircraft tested, the rudder was located above the roll axis whereby the rolling moments produced by rudder deflection were in a direction to reduce that due to dihedral effect.

The yaw-pitch coupling case has occurred in hover for some VTOL configurations such as the SC-1 (ref. 33) where a common yaw and pitch nozzle is used for control. An arbitrary upper limit of 2° pitch angle change was set to reduce the loss in precision of control when full yaw control inputs are made. No quantitative limit is specified for STOL operation; however, a pitch-down direction is preferred to reduce the tendency to approach a pro-spin condition that may occur if high angle of attack is induced.

The cross-coupling in the vertical direction must be minimized to provide good hover characteristics in precision tasks such as rescue or crane operations. Aircraft with demand bleed reaction control systems, as used on the P-1127, inherently have this coupling problem since engine thrust will be reduced proportional to the magnitude of bleed air extracted. The upper limit of 0.05 g can result in appreciable loss in altitude when large heading changes are made and when only small excess thrust margins are available. The value of 0.05 g corresponds to the lower limit of vertical thrust control power required (see section 4.5).

3.17 DIRECTIONAL CHARACTERISTICS IN STEADY SIDESLIPS

The criteria in section 3.17 define directional stability in forward flight in terms of yaw control deflection with sideslip angle. This interpretation of directional stability is that normally used by the

pilot. The linearity of control force and deflection with sideslip is required only over the practical design range of ±15°, after which nonlinear characteristics are allowed. A further relaxation in linearity is intended for air speeds less than 30 knots where much larger sideslips are obtained in cross winds and where aerodynamic forces and moments are less significant. For example, the P-1127 (ref. 7) exhibits static directional instability in hover because of induced flow effects and inlet drag; yet these characteristics, although not desirable, are not considered unacceptable by pilots. Another point to consider is that directional stability alone will not guarantee satisfactory aircraft behavior in low-speed flight since, as previously noted in section 3.16, cross-coupling effects can cause severe problems even though an aircraft is directionally stable. Further, too much aerodynamic directional stability can be detrimental because of increased sensitivity to gusts (see discussion for STOL aircraft, ref. 4).

3.18 SIDE-FORCE CHARACTERISTICS IN STEADY SIDESLIPS

This criterion for a proper sense of bank angle with sideslip has traditionally appeared in handling qualities documents for many years. The primary purpose is to help the pilot sense the direction and magnitude of sideslip angle, which becomes more important in IFR operation. Although there is no known evidence to support the theory that zero bank-angle variation with sideslip is acceptable, flight tests of STOL aircraft (refs. 4 and 52) indicate the desirability of a low gradient. This was helpful in cross-wind operation where a wing-down approach technique could be made close to the ground with less concern for striking the wing tip since only a small bank angle was needed to compensate for a large sideslip angle. Further, evidence of the desirability of a linear sideforce relationship was indicated in the flight tests of a STOL seaplane (ref. 53). In this case, the side-force characteristics apparently changed in magnitude as the aircraft approached the water to land, requiring a heading correction to be made by the pilot.

Additional information from a systematic investigation of side-force characteristics is needed to establish this criterion more firmly.

3.19 LATERAL-DIRECTIONAL DYNAMIC STABILITY

The criteria of section 3.19 (ref. 3) require that any roll-yaw oscillations superimposed on the normal control mode due to a disturbance input should exhibit at least the frequency damping characteristics shown in figure 97 over the speed range specified. Also, there should be no tendency for perceptible small-amplitude oscillations to persist or for pilot-induced oscillations to result from the pilot's attempts to perform the required tasks.

After a failure of a SAS, minimum damping characteristics should be at least those of the single failure curve in figure 81. Small-amplitude residual oscillations are permitted, provided they are not objectionable to the pilot.

These criteria were based primarily on data obtained from an investigation conducted by the Canadian National Aeronautical Establishment using a variable stability helicopter and a "model-controlled" method of simulation. These flight trials were conducted concurrently with the investigation of longitudinal dynamic stability described in section 2.8 of this report.

Figure 82 shows the damping ratio (ζ), natural period (T_n), and pilot rating for test conditions in the hovering task. The shaded curves represent faired 3-1/2 and 6-1/2 pilot rating boundaries. The solid curves represents straight-line fairings of the shaded curves. The broken curves represent the old (ref. 1) AGARD 408 normal and single failure limits. The new boundaries (solid curves) are replotted in figure 83. Finally, these new boundaries were replotted in the S-plane form to produce figure 81. The net result is that for the hovering task the normal dynamic stability limit (3-1/2 pilot rating) is basically unchanged, whereas the damping ratio required for the single failure limit (6-1/2 pilot rating) was reduced by a factor of 0.1.

Figure 84 (taken from ref. 4) shows the directional stability and damping characteristics of several STOL aircraft. The data on the left-hand portion of this figure show no correlation between pilot opinion and directional frequency. For these aircraft the behavior was dominated by the damping and cross-coupling associated with low stability (low directional frequency). The right-hand portion of figure 84 relates pilot opinion and the Dutch-roll damping parameter, $\zeta\omega_d$, for STOL aircraft with a directional frequency range of 0.5 to 1.2 rad/sec. For damping ratios less than 0.3, $\zeta\omega_d$ is approximately inversely proportional to time to half or double amplitude of the Dutch-roll oscillation. In general, the ratings improve as the damping is increased; however, turn coordination also influences the ratings to a considerable degree, as indicated by the less favorable ratings shown in the right-hand figure when $\Delta\beta/\Delta\phi$ is larger (0.6).

The lateral-directional dynamic stability criteria of section 3.19 (ref. 3) may be too lenient for the STOL approach condition in that a damping ratio of at least 0.3 may be needed for the lower frequency (long-period) oscillations.

Section 4.0

HOVERING AND VERTICAL FLIGHT-PATH CHARACTERISTICS

4.1 GENERAL

The criteria for hovering and vertical flight-path characteristics presented in section 4 of reference 3 are based primarily on the results of four height control studies and to a lesser extent from tests of V/STOL aircraft. Important considerations include: (1) ground effect, particularly in regard to performance, (2) vertical flight-path control in terms of control power, response time, and cross-coupling; (3) hovering precision; (4) vertical thrust margins; and (5) vertical velocity and thrust response.

4.2 CHARACTERISTICS IN GROUND INTERFERENCE REGION

The ground effect criteria presented in section 4.2 (ref. 3) are basically unchanged from the original criteria of reference 2 except interference effects experienced during STOL operation are included.

In references 5 and 46, it was shown that interference effects could severely affect aircraft performance. Shown in figure 85 for the XC-142 is the area of the flight envelope that had to be avoided because of deterioration in controllability and performance in ground proximity. To avoid this area, it was necessary to initiate all transitions from hover at a height above the ground greater than 25 ft. It then became necessary to consider sink rates that would result from the loss of one engine and the design limits of the landing gear. A large payload penalty resulted when height-velocity and handling qualities requirements were met.

The criteria of section 4.2 also require that the aircraft designer consider powered controls to isolate the hovering controls from disturbances experienced by the aerodynamics surfaces (e.g., buffeting in ground effect or unbalance in tail-to-wind flight). It may be desirable, therefore, to have powered controls even though they may not be dictated by characteristics of the conventional flight regime.

4.3 VERTICAL FLIGHT-PATH CONTROL

The criteria of section 4.3 (ref. 3) state that for satisfactory flight-path control during all phases of STOL flight operation below V_{con} (including approach, landing flare, touchdown, and waveoff), the vertical aircraft response characteristics obtained at a constant attitude resulting from any combination of inputs from throttle, collective, and thrust vector controls should meet the values listed in table 4.1.

Table 4.1

Minimum Vertical Flight-Path Control Characteristics in STOL Operation

Item	Mode*	Parameter to be measured	Level for satisfactory operation	Minimum level for acceptable operation
Control power	A	Incremental normal acceleration	±0.1 g	Insufficient data
	B	Incremental normal acceleration	±0.1 g	Insufficient data
	C	Steady-state climb angle	6° or 600 ft/min	200 ft/min
	All	Incremental descent angle	2° greater than selected approach angle	Insufficient data
Response time	A	Aircraft response	Achieve mode XA in less than 0.5 sec	Insufficient data
	B	Aircraft response	Achieve mode XB in less than 1.5 sec	Insufficient data
	C	Aircraft response	Achieve mode XC in less than 2.0 sec	Achieve mode XC in less than 4.0 sec
Cross-coupling	All	Pitching moment	Not objectionable	Not objectionable

*Mode A: For flare and touchdown control when less than 0.15 g can be developed by aircraft rotation using pitch control alone.
Mode B: For flight-path tracking when more than 0.15 g but less than 0.30 g can be developed by pitch control alone.
Mode C: For gross flight-path changes regardless of the normal acceleration developed by pitch control.

Vertical control of flight-path angle during approach, flare, and touchdown, and during rotation and climbout is an important consideration for STOL operation because of the effect on field length performance. The ability to routinely operate satisfactorily from short fields with obstacles in the approach and climbout paths depends on having precise control of flight-path angle. It could be expected that greater difficulty would occur in the control of STOL aircraft because STOL operation normally takes place on the backside of the power-required curve where control of flight path by attitude change results in a speed

divergence. As noted in reference 4, however, backside operation per se has not proved to be a major problem for the aircraft tested. There is reason to believe, however, that systematic studies should be conducted to arrive at a criterion for all types of aircraft.

During low-speed flight, sufficient normal acceleration may not be available by pitch control alone for vertical flight-path control for STOL operation of V/STOL aircraft, and the pilot must use additional methods of developing normal acceleration. Powered lift for flight-path control can be considered in three general modes: (1) controlling rate of sink at flare and touchdown; (2) acquiring and tracking a particular flight-path angle during approach; and (3) making gross changes in flight path for waveoff and turning flight. Satisfactory performance of the foregoing tasks depends on the amount of normal acceleration available from powered lift, the aircraft response time, and the degree of cross-coupling. The values desired by the pilot depend on how critically the particular flight mode must be controlled. For example, altitude control during flare and touchdown must be precise and requires a short response time for this critical maneuver. It is equally important that cross-coupling effects between powered lift and aircraft rotation be minimized so that the pilot can precisely adjust rate of sink and aircraft attitude independently as required for optimum landing and takeoff performance.

Consideration of these points is given in the criteria presented in table 4.1. As noted, different modes of operation are specified for STOL operation of V/STOL aircraft, depending on the precision of flight-path control required. As expected, the pilot desires increased vertical response time and "g" from power as he gets closer to ground contact. To determine whether the criteria for Mode A or B apply, abrupt longitudinal control steps should be performed at the appropriate trimmed flight-path angle. Compliance with the criteria should be demonstrated by performing steps with the flight-path control device with the aircraft attitude maintained constant with the pitch control. Mode C applies equally to all aircraft regardless of the means to produce the response.

In tests of the Br 941 aircraft (ref. 25), engine response to small throttle changes had a 0.5-sec lag plus a first-order time constant of 0.7 sec. There was no appreciable lag between vertical "g" and power changes (i.e., no aerodynamic, slip-stream lag). It was possible with throttle alone to obtain more than ±0.1 g, which resulted in satisfactory flight-path tracking down to about 50 ft. The pilot felt that longer engine time lags and time constants would have degraded his ability to track the ILS glide slope. This response was not adequate when using power to arrest the sink rate at touchdown. In general, none of the STOL aircraft tested thus far (ref. 4) were flared for touchdown by using engine thrust for several reasons: (1) engine response was too slow; (2) the aircraft had to be rotated for proper ground attitude; and (3) power changes produced undesirable changes in air speed. As a result, "g" was obtained, as for conventional aircraft, by rapidly increasing aircraft attitude which also increased angle of attack. The touchdown maneuver for STOL aircraft is, of course, similar to the height control problem for VTOL aircraft in terms of precision. In this respect, values of overall thrust response should not be greater than 0.5 sec and 0.1 g should be available. The response for gross changes in flight path (away from the ground) are less stringent; for example, a 2.0-sec delay was judged satisfactory (ref. 4).

Admittedly, the vertical flight-path criteria in their present form are weak and more firm quantitative values are needed for both control power and thrust response. As is true for control of other axes, cross-coupling effects and interrelated items have an effect on the pilot's assessment of precision of control. Included are the following:

(1) Static longitudinal stability
(2) Short-period and phugoid frequency and damping
(3) Direct lift control
(4) Effect of automatic power compensation
(5) Ground effect on lift, drag, and pitching moment
(6) Gust sensitivity (lift curve slope)
(7) Power "backsidedness"
(8) Trim change with power (magnitude and direction)
(9) Thrust and control system response (lags)

As can be envisioned, a systematic evaluation for the foregoing items is a formidable task and it is difficult to generalize on answers from specific aircraft because of the inability to vary the significant parameters over wide enough ranges. Studies have been made of most of these items separately (e.g., see ref. 53 for item 8); however, an overall evaluation still remains to be carried out.

4.4 HOVERING PRECISION

Originally (in ref. 2), the criteria were divided into two sections, one entitled "Height Control" and the other "Hovering Precision." The first required control of vertical speed to ±1 ft/sec, the second required that any chosen point on the aircraft remain within a circle of 3 ft radius. One requirement specified speed control and the other specified position control. The new criteria of section 4.4 (ref. 3) specify horizontal and vertical position because the overall objective is to assure that a VTOL aircraft can operate satisfactorily from a confined space or with a sling load.

Section 4.2, Characteristics in Ground Effect, specifies handling qualities while in ground effect at any speed, while section 4.4 specifies a hover positioning capability. The two sections, therefore, complement each other and are not redundant.

Insufficient flight-test data exist to determine if the present generation of VTOL aircraft have met the old specifications or can meet the new criteria.

Aircraft capable of lifting an external sling load must have a hovering precision capability that will enable the pilot to (1) position the aircraft horizontally over the load and (2) position the aircraft vertically over the end of the cable. The length of the cable will determine two things. First, the longer a cable, the smaller are the upset moments that will be generated from a given off-center displacement.

Second, as height increases, ground reference cues decrease. For these reasons, the horizontal position requirements for the out-of-ground effect condition are less stringent than for the in-ground effect condition.

Other VTOL aircraft, such as the VTOL fighter, have their own peculiar hover precision requirements that are generated by the takeoff and landing environment rather than by the fighter mission. Consider, for example, the hover precision requirement generated when a large number of VTOL fighters operate from a small clearing. These aircraft will have a hover precision requirement best described as a zero airspeed formation flying task.

VTOL aircraft will be required to hover in extreme wind conditions. Care should be exercised to ensure that vehicle design is not overly complicated to meet one specific requirement. For example, to require all VTOL aircraft to hover with a 35-knot tailwind would penalize some aircraft, especially those that do not have vectored thrust capability. For this reason a designated wind condition is not specified in the criteria and should be determined by the procuring agency.

As noted in the first paragraph of section 4.4, the criteria specify a horizontal and vertical position because establishing and maintaining a position are the primary objectives. The control characteristics that enable an aircraft to meet the hovering precision criteria are a function of many parameters such as thrust-to-weight ratio, control sensitivity, velocity damping, control time lags, and control free play. The horizontal (lateral and longitudinal) parameters are adequately covered in other sections of this report. Documentation for the vertical parameters for height control are presented in the following sections.

The parameters that affect height control have both individual importance and interdependence and vary with hover height and the nature of the ground effect. Negative ground effect could amplify the importance of thrust-to-weight ratio and control sensitivity. A rapid onset of negative ground effect would probably increase the effect of any control time lags (including engine response times) and normal velocity damping. Insufficient flight-test data are available to document this, but the relative importance of normal velocity damping, time lags, and thrust-to-weight ratios is shown in figure 86 (taken from ref. 54). The same three parameters plotted versus pilot opinion are presented in figure 87 (taken from ref. 55). Inspection of figure 87 indicates a deterioration of pilot rating with increased time delay for all cases tried and this agreed with data presented in figure 86. Upon further inspection, it is noted that an increase in time delay results in a greater rate of control deterioration for the condition of high thrust-to-weight ratio, either with or without damping, than for a low thrust-to-weight ratio. It must be realized that low thrust-to-weight ratios limit the amount of vertical maneuvering a pilot will use. The smaller the control input, the less tendency there is to set up large sink rates and therefore less adverse effect will be realized from long time delays. Precision hovering and rapid maneuvers require high thrust-to-weight ratios and short time lags. Also, a proportional increase in the amount of vertical velocity damping is required to prevent overcontrolling and pilot-induced oscillations.

Finally, it should be noted that criteria for hovering precision are closely related to the criteria of the next two sections, namely, vertical thrust margins and vertical thrust response. That is, the required thrust-to-weight ratio for takeoff and landing is specified in section 4.5 and the thrust response criteria, including first-order time constants, are specified in section 4.6. More specifically, figure 86 presents pilot rating curves for different combinations of thrust to weight, vertical velocity damping, and time constant. In conjunction with figure 86, the following is quoted from reference 54. "A minimum damping level requirement is established as control power is increased. The $\tau = 0.07$ also defines a minimum satisfactory control power level of about 1.06 at a damping level of -0.5/sec. Increasing the damping at this control power causes sluggish or velocity limited (less than 3 ft/sec) operation while decreased damping causes overcontrolling. Increasing the time constant to 0.37 sec shifts the minimum control power requirement to the right (1.08 g) and raises the minimum damping requirement to about 0.2/sec. Raising the time constant to 0.87 sec drastically increases the minimum damping requirement and makes damping insensitive to control power changes in the region tested."

The time constant specified in section 4.6, Thrust Response, is 0.3 sec, and figure 88 shows the damping level of current VTOL aircraft.

4.5 VERTICAL THRUST MARGINS

The original criteria (ref. 2) specified a thrust-to-weight ratio of 1.05 for takeoff and 1.15 for landing, but did not consider inherent vertical velocity damping. The new criteria (fig. 89) are based on data from two height control studies that considered vertical thrust margins as a function of inherent vertical velocity damping which helps check a rate of descent. The first investigation (ref. 56) utilized a variable stability helicopter while the second (ref. 54) used a vertical height apparatus (simulator) that had a 100-ft vertical height capability with real-world motion cues.

The term "inherent vertical velocity damping" is stressed here because artificial velocity damping, especially in the landing condition, may reduce the thrust-to-weight ratio available to the pilot. Artificial velocity damping will increase hover steadiness but will not necessarily increase the vertical thrust margin required; for example, when a vertical velocity demand system is used. As reported in reference 5 however, height damping gave the effect of reducing thrust-to-weight ratio available since the height SAS would reduce or "wash out" some of the pilot input. The criteria of section 4.5 were established as a function of inherent vertical velocity damping only and if artificial vertical damping is needed, consideration must be given to the effect on thrust-to-weight ratio. For the remainder of this section, inherent vertical velocity damping is implied.

Data from variable stability helicopter tests (ref. 56) show that the takeoff thrust-to-weight requirement (fig. 89) is more demanding than the landing thrust-to-weight requirement (fig. 90) which is the

opposite of previous criteria. With zero damping, the takeoff and landing vertical thrust margin requirements are nearly equal. The big difference, however, is that the takeoff thrust-to-weight ratio must increase as damping increases while the opposite is true for the landing condition.

The takeoff criteria of section 4.5 consider two conditions. The first, in-ground effect, permits aircraft with negative ground effect (shaded area) to initiate a takeoff with a thrust-to-weight ratio as low as 1.05 and with velocity damping equal to or less than -0.16. A thrust-to-weight ratio of 1.05 is necessary to get the aircraft through the adverse ground effect region in minimum time and is the only takeoff condition covered by the original criteria (ref. 2). The second, out-of-ground effect, requires a thrust-to-weight ratio equal to or greater than 1.1 and a damping value equal to or less than -0.325. These vertical thrust margin criteria, established from reference 56 results, were based on the need to accelerate quickly from hover to forward flight in a climbing turn with a rate of climb equal to or greater than 600 ft/min.

It may appear that the greater vertical thrust margin requirements for takeoff would make the lower values for landing unnecessary. However, two separate requirements for takeoff and landing have an advantage in the case of a STOL takeoff followed by a VTOL landing. In another example, it permits an aircraft to take off at the design gross weight, using afterburners, and immediately land, without afterburners, at approximately the same gross weight but with a lower thrust-to-weight ratio if sufficient velocity damping is available.

The landing vertical thrust margin criteria presented in section 4.5 were established primarily from the results of reference 56. It was concluded that when considering only the approach task, including a flare and landing, satisfactory operation is possible for thrust-to-weight ratios as low as 1.03 if the vertical velocity damping is equal to or greater than -0.25/sec. In both references 54 and 56, the primary factor considered by the pilots when evaluating vertical thrust margins in the landing maneuver was their ability to check a given rate of descent. As stated in reference 54, the pilots limited the rate of descent whenever control power and/or damping were decreased. Pilot comments indicated that 10 ft/sec was used as an average comfortable rate of descent when combinations of control power and damping permitted. The time required to check a given rate of descent increases as the thrust margin and/or damping decreases and it becomes harder for the pilot to estimate when he should start checking the rate of descent. The times required to check a 10 ft/sec rate of descent with zero control lag and the permissible combination of thrust-to-weight ratio and velocity damping of section 4.5 are presented in table 4.2. Pilot comments indicate that times greater than approximately 5 to 6 sec are unacceptable.

Table 4.2
Vertical Height Deceleration Characteristics

Maximum T/W ratio available	Inherent vertical velocity damping	Time required to check a 10-ft/sec rate of descent, sec
1.10	0	3.1
1.05	-0.10	4.8
1.03	-0.25	5.1

Note: Results based on zero control system lag.

Several operational factors must be considered when establishing compliance with vertical thrust margin criteria. First, for jet-lift VTOL aircraft, takeoffs and landings may have to be made quickly to avoid thrust loss that can occur after a local area is heated by exhaust efflux. Second, for some V/STOL concepts, power management may cause problems for the pilot and he may not be able to readily use the total T/W ratio available. Aircraft that use both lift/cruise engines and lift engines usually provide separate throttles for each set of engines. To take off, the pilot may first advance one throttle to the full-power position, adjust the thrust vector lever to minimize ingestion, and then use the second throttle as necessary to accomplish the liftoff and climb. To check a rate of descent during the landing maneuver, the pilot may not be able to manipulate two sets of throttles and adjust thrust vector angle as expeditiously as desired. Another factor to consider is the performance loss when lift engines are used for pitch, roll, or yaw control.

4.6 VERTICAL VELOCITY AND THRUST RESPONSE

The vertical velocity and thrust response criteria presented in section 4.6 were established by considering four basic parameters: (1) thrust-to-weight ratio, (2) control sensitivity, (3) vertical velocity damping, and (4) control system time lag. Vertical velocity response characteristics are judged on the basis of a small (1-in.) abrupt input of the cockpit height control typical of normal piloting techniques.

The maximum and minimum changes in vertical velocity were established by first considering control sensitivity, Z_δ. Data in figure 91 (ref. 56) show a satisfactory range from 0.05 to 0.4 and an optimum control sensitivity range of 0.1 to 0.2 g/in. with a vertical velocity damping of -0.25/sec. The following is taken directly from reference 56. "At lower sensitivities, below 0.1 g/inch, the pilots complained of excessively large control motions and difficulty in establishing the 1 g thrust condition for hovering. For sensitivity values above 0.3 g/inch the pilots felt it was necessary to exercise

extreme care in order to avoid overcontrolling, which, in some systems, could lead to overstressing the lift-system dynamics components. The maximum value simulated, 0.6 g/inch, was sufficiently sensitive to give the pilot the impression of a pressure control; that is, the control motions were sufficiently small to be imperceptible to him."

After control sensitivity was determined, the next step was to determine the range of vertical velocity damping that could be expected in current generation VTOL aircraft. Figure 87 shows that current VTOL aircraft have a vertical velocity damping value of approximately -0.1 to -0.2/sec and only light helicopters exceeded a value of -0.5/sec.

Vertical thrust response criteria, therefore, should cover a range of control sensitivity between 0.1 and 0.4 g/in. and a range of vertical velocity damping between 0 and -0.5/sec. Figures 92 and 93 show the effect of vertical velocity damping on thrust response with zero lag for control sensitivities of 0.1 and 0.4 g/in., respectively. The criteria limits are taken as the extreme conditions of these two parameters, that is, maximum control sensitivity and zero damping or minimum control sensitivity and high damping. Figure 94 presents, on expanded scale, the extremes of these two conditions. From these curves, a maximum and minimum thrust response was established 1 sec after a 1-in. control input. Reading the curves directly would establish a maximum vertical velocity of 772.6 ft/min and a minimum of 152 ft/min. These were rounded off to 775 and 150 ft/min and used for the criteria of section 4.6.

The criteria of section 4.6 also state that 63 percent of the above response must be obtained within 0.5 sec. This criterion was established from data presented in reference 54 (fig. 95). This value of the first-order time constant was experimentally determined with zero velocity damping and near-optimum control sensitivity.

Section 5.0

TRANSITION CHARACTERISTICS

5.1 GENERAL

Good transition characteristics are essential for successful use of V/STOL aircraft for a number of reasons. First, it may be desirable for fuel economy reasons to perform a transition quickly and to minimize the time spent in the terminal area. Second, transitions must be performed in critical phases of flight, such as the climb and/or landing approach, when the pilot must be able to maintain precise control of the flight path, particularly for the IFR operation. Finally, transitions necessarily occur at a time of high pilot work load, when configuration changes (such as selection of landing gear and flaps, starting the lift engines, etc.), communications and navigation duties must be carried out. For the most part the various V/STOL aircraft evaluated thus far have had satisfactory transition characteristics for VFR operations. Additional information is needed to set final criteria for IFR operation. Discussion in the following paragraphs cover the information used to establish those criteria for handling qualities that govern aircraft behavior in going from powered lift flight to aerodynamic lift regime, and vice versa, for both VTOL and STOL aircraft.

5.2 ACCELERATION/DECELERATION

For VTOL aircraft, the criteria of section 5.2 (ref. 3) state that it should be possible to decelerate rapidly and safely at constant altitude or in a descent up to the maximum approach angle required by the mission, to acquire and maintain both shallow and steep flight path angles, and to stop quickly and precisely over a preselected hover spot. In some areas the criteria suggest quantitative values, where depending on the mission, acceleration and deceleration values up to 0.5 g in level flight are desired. In addition, in order to reduce recirculation effects, it is desirable to be able to accelerate continuously from a rolling takeoff (RTO) to V_{con} and decelerate smoothly to a rolling landing.

For STOL aircraft, the criteria state that it should be possible to accelerate from V_{app} to V_{con} in level or climbing flight, to decelerate quickly from V_{con} to V_{app}, and to acquire precisely and maintain both shallow and steep flight path angles.

The purpose of these criteria is to ensure that the aircraft is capable of going from powered lift flight to aerodynamic lift flight and vice versa, without large changes in aircraft attitude, angle of attack, airspeed, and trim-factors that would compromise the pilot's ability to fly the aircraft accurately along a chosen flight path in all environmental conditions. In addition, the aircraft should have the capability to decelerate as needed at any portion of the speed range to attain quickly a particular approach speed or to avoid overshooting a desired touchdown area.

The time required for making a transition must be established by the particular mission; however, it is necessary from safety considerations that the rate desired by the pilot should not be governed by limitations in controllability about any axis.

Operation of various VTOL and STOL aircraft has indicated that the V/STOL concept itself may have certain built-in limitations on acceleration/deceleration handling characteristics. Further, these characteristics vary depending on the direction of transition. The following discussion shows the effect of thrust vector controls on the acceleration/deceleration characteristics of VTOL aircraft and how the VTOL concept in turn determines the type of thrust vector control used. In this respect, attention should also be directed to section 1.8, "Characteristics of Thrust Vector Controls".

The P-1127 aircraft was equipped with a proportional-position thrust vector control lever that changed the direction of the engine thrust. The magnitude and direction of the aerodynamic vectors are controlled indirectly through changes in aircraft attitude. The pilot, therefore, can change the magnitude and direction of the engine thrust vector independently of the aerodynamic vectors. As shown in reference 7, the rate at which the P-1127 pilot moved the proportional thrust vector control was directly related to the magnitude of the vector. When a large engine thrust vector was rotated, as during takeoff (fig. 96), a rate of approximately 4 deg/sec was selected. (Note that a maximum rate of approximately 175 deg/sec is available with this system.) This operation provided an initial acceleration of approximately 0.2 g and an overall average acceleration (0 to 160 knots) of 0.43 g. This accelerating transition was initiated from a 60 ft hover altitude with a +7 deg (nose-up) pitch attitude. During the transition the altitude increased approximately 40 ft and the pitch attitude increased to a maximum of +12 deg at 90 knots and then decreased to +5 deg at 160 knots. A higher rate of thrust vectoring could have produced higher accelerations but also a possible loss in altitude, neither of which would necessarily be desirable. A thrust vectoring rate of only 4 deg/sec was therefore satisfactory for this particular accelerating transition maneuver.

Later experience has indicated that it is desirable to start with a positive rate of climb so that the thrust can be vectored more rapidly. Vectoring rate has increased to 8 to 10 deg/sec and there is usually a reduction in pitch attitude to approximately zero, returning through +7 deg by about 50 knots.

During a decelerating transition (fig. 97), the pilot commanded a thrust vectoring rate of approximately 45 deg/sec. This much larger rate was possible because of the small magnitude of the engine thrust vector. This decelerating transition was initiated at 160 knots with a +6.5 deg pitch attitude and a low engine power setting. From 160 to 80 knots, the pitch attitude varied ±2 deg while the power was increased to produce a maximum deceleration of 0.46 g. At 80 knots, the thrust vector was rotated from the 5 deg forward position to the vertical position, after which the aircraft pitch attitude was increased to +14 deg to hold altitude and decelerate from 80 knots to hover with an average deceleration of approximately 0.2 g. The maximum variation in altitude during this transition was 50 ft.

In tilt-wing aircraft, such as the CL-84, the aerodynamic vectors are rotated with the engine thrust vector. The pilot must thus command a thrust vectoring rate that is compatible with both the magnitude of

the aerodynamic vectors and the engine thrust vector. This operation is further complicated by the fact that the maximum thrust vectoring rate available is a function of wing angle and direction of thrust rotation. The CL-84 wing could be rotated up at a rate of 6 deg/sec. The maximum downward rate of 12 deg/sec was linearly decreased to 2.6 deg/sec between wing angles of 45 and 5 deg. The pilot, therefore, did not have direct control of thrust vectoring rate as his control was only an on/off switch. Variations in thrust vectoring rate could be achieved by intermittently turning the switch on and off.

An accelerating transition time history for the CL-84 is shown in figure 98 (taken from ref. 6). The initial conditions for this maneuver were hover OGE, zero pitch attitude, and an 86 deg wing angle. The transition was initiated with a vector rate of approximately 7 deg/sec, thus producing an initial acceleration of 0.2 g. After a brief (2 sec) interruption in the thrust vectoring command, the pilot used the maximum thrust vectoring rate of 10 deg/sec for the remainder of the transition, resulting in a maximum acceleration of 0.44 g. In this accelerating transition, the initial aerodynamic vector was small, permitting the pilot to use a high thrust vectoring rate without experiencing control coordination problems. The CL-84 pilot's inputs, therefore, were similar to those used with the P-1127; that is, a large engine thrust vector was rotated in such a manner as to give nearly identical acceleration characteristics. Decelerating transitions of tilt-wing VTOL aircraft are different than jet lift aircraft because the pilot is required to tilt a large aerodynamic vector. This operation creates a control coordination problem because the pilot must select a wing tilt rate that is compatible with the aerodynamic vector and the engine thrust vector. As stated in reference 6, this completely unfamiliar technique was difficult to perform; it was further complicated by the need to operate the wing tilt switch intermittently in order to get a variable rate to match the lift required. In order to adjust rate of deceleration, the pilot was required to change both power and wing tilt rate in a coordinated manner; consequently holding the deceleration at any fixed rate was very difficult. A CL-84 decelerating transition time history (fig. 99) shows that the pilot commanded a thrust vectoring rate of 3 deg/sec for the major portions of the maneuver (wing angles 15 to 60 deg) and then commanded a maximum available rate of 6 deg/sec for the remainder of the transition (60 to 86 deg). This technique produced a nearly constant deceleration of 0.15 g. Although the aircraft is capable of higher decelerations, the pilot's control coordination problems become greater.

The X-14 aircraft has unique characteristics during the transition that make a high thrust vectoring rate desirable. The X-14 aircraft has limited thrust vectoring capability because the engine thrust vector can only be rotated 68 deg within the range of 22 to 90 deg from the horizontal (22 deg cruise position, 90 deg hover position). At cruise and hover, the thrust vectors are parallel to the X-Z plane of the aircraft, but at intermediate angles a lateral displacement of the thrust vector occurs due to the peculiar rotation of the thrust diverter system. This lateral displacement of the thrust vector produces two undesirable results. First, it reduces the vertical component of the thrust vector without a complementary increase in the horizontal component. Second, this lateral displacement affects the airflow at the trailing edge of the wing root producing unsteady flight (rolling oscillations).

A typical X-14 accelerating transition is accomplished in the following manner. From hover, the acceleration is initiated by a nose-down pitch attitude change. The high incidence angle of the wing (11 deg at the root) makes it possible to maintain the initial nose-down pitch attitude throughout the transition and still develop sufficient aerodynamic lift for level flight. Thrust vectoring is usually not initiated until a forward speed of 23 ft/sec is achieved because below this speed, there is insufficient elevator control power to counteract the nose-up pitching moments generated by the induced flow effects at intermediate thrust vector angles. As soon as sufficient aerodynamic lift is developed, engine power is reduced and the engine thrust vector (of decreased magnitude) is rotated at a maximum rate of 15.8 deg/sec to the cruise position of 22 deg in approximately 4 sec. A higher thrust vectoring rate would be desirable to minimize the effect of undesirable flow characteristics at the intermediate thrust vector angles. During decelerating X-14 transitions, airspeed is reduced initially by reducing engine power; the thrust vector is then rotated to the vertical position at a maximum rate of 15.8 deg/sec, requiring approximately 4 sec. A higher thrust vectoring rate for both the accelerating and decelerating transitions would be desirable, but a means for obtaining a variable rate would be required so that the pilot could accurately make small changes in diverter position while hovering.

For the XV-5A aircraft at low speed, the wing fan exit louvers are used for speed (thrust vectoring) control as well as for height control, roll control, and yaw control. In addition, the angle (thrust vectoring angle) of the louvers determines the amount of fan roll control power available to the pilot. Fan roll control is phased out as a function of louver angle, as speed and aerodynamic aileron control increase. The pilot, therefore, could prematurely phase out fan roll control before adequate aileron roll control was established by commanding a thrust vectoring rate that is too high. As reported in reference 9, specific attention was required to insure that a "rule-of-thumb" relationship of 2 knots of airspeed for each degree of louver angle was maintained. If this 1:2 relationship was exceeded prior to 40 knots, a loss of lateral control power was observed. A high degree of pilot attention was required to maintain the louver angle-airspeed schedule (pilot rating of 5). The maximum thrust vectoring rate built into the XV-5A aircraft was 3 to 4 deg/sec. A time history presented in figure 100 (taken from ref. 9) shows that during an accelerating transition from hover, the pilot commanded an overall average thrust vectoring rate of 1.6 deg/sec and an acceleration of 0.13 g. The time required to accelerate from zero to 40 knots was approximately 16 sec.

The wing tilt rate available on the XC-142 aircraft was a function of wing angle and the direction the wing was moving. A maximum tilt rate of 9 deg/sec was obtained only between wing angles of 35 and 40 deg. Between wing angles of 10 and 80 deg, a maximum wing tilt rate of 6 to 9 deg/sec was available. From 10 to zero deg and 80 to 100 deg, the maximum thrust vectoring rate available decreased in a nonlinear manner to near zero rate at the extremes. Pilot control of thrust vectoring (wing angle) was accomplished by a variable rate switch located on each collective control or by a constant rate switch on each number 4 throttle. When using the constant rate switch, a rate of only 3 to 4 deg/sec for all wing angles could be obtained. As reported in reference 5, an accelerating transition was accomplished by using the variable rate switch on the collective and commanding the maximum rate available. It was reported that an accelerating transition from hover to 140 knots in 12 sec was easily accomplished (pilot rating of 1). To satisfy this condition, it would be necessary to have an average thrust vectoring rate of 7.15 deg/sec (86 deg in 12 sec) and an average acceleration of 0.61 g. A time history of an acceleration transition is shown in figure 101 (ref. 5) in which the pilot did not initially (0-20 knots) command the maximum available thrust vectoring rate, but

did use a 5 deg nose-down attitude change. From 20 knots to 80 knots, the pilot commanded the maximum available thrust vectoring rate but returned the aircraft pitch attitude to zero deg. This technique resulted in an average thrust vectoring rate of 3.6 deg/sec and an average acceleration of approximately 0.1 g from 0 to 20 knots, and 0.29 g from 20 to 80 knots. The time required to go from 0 to 80 knots was approximately 22 sec. This performance, in effect, was only one-fourth of the performance capability reported qualitatively by the pilots and the report does not reconcile these differences.

Other VTOL concepts may have unique characteristics that determine maximum or minimum thrust vectoring rates. For example, tilt jet engine concepts may experience inlet airflow distortion problems that will limit the maximum thrust vectoring rate.

The foregoing documentation was presented to show some of the factors that determine or establish thrust vectoring rates. Sufficient data are available to show that one minimum or maximum rate will not satisfy all VTOL concepts, but insufficient data are available to establish a satisfactory rate for each VTOL concept. In addition, the foregoing data show the desirability of providing the pilot with a variable rate thrust vector control. Further, the limitations for IFR operation have not been defined. It is expected that only relatively low deceleration values will be used to reduce pilot workload in the landing approach task. Early experience with the DO-31 aircraft (ref. 47) indicated that deceleration values of 0.07 g were used to provide sufficient tracking time on the ILS to assess the approach such that confidence was gained to proceed to the landing. Finally, real life operation is needed to assess the passenger comfort aspects of transition for civil use.

Other acceleration/deceleration characteristics that determine the rate of transition commanded by the pilot are the apparent trim changes during the maneuver. Most V/STOL aircraft equipped with a SAS attempt to counteract trim changes by programming appropriate moment-producing devices as a function of thrust vector angle. If properly programmed, these aircraft should have no apparent trim changes during transition and should therefore have minimum effect on the pilot's ability to effect a transition. This feature makes it desirable to use thrust vectoring rather than aircraft attitude change for low speed maneuvering, because the pilot is not required to manage large trim changes. (An exception to the use of this technique is with the X-14 aircraft, where thrust vectoring cannot be used for low speed maneuvering because of the large pitching moments generated by thrust vectoring.)

5.3 FLEXIBILITY OF OPERATION

The need for changing the direction of transition at will is based on flight safety and operational considerations. Aircraft which have too narrow a transition corridor or those which rely on automatic devices to accomplish a safe transition are more prone to cause problems. Examples where troubles have been encountered on experimental V/STOL aircraft in transition include the VZ-3 deflected slipstream aircraft and the XV-5A fan-in-wing concept. For the VZ-3 aircraft, operational experiences obtained during transitions are discussed in references 57 and 58. The transition corridor, defined as the airspeed range between the maximum allowable speed for a given flap setting (based on structural considerations) and the stall speed, was relatively small for this aircraft (as small as 5 knots and less at high-flap deflections). The flap placard speed was relatively low because of the desire to reduce weight to provide hover capability. In attempting a transition, the effect of this narrow corridor was seen in the lack of flexibility of aborting the transition. In this case, control of the aircraft was lost when the pilot reduced power to arrest a high climb rate. In another example, a transition from high speed was being attempted with the XV-5A aircraft. The transition procedures (described in ref. 9) for this aircraft involved the operation of a switch which automatically programmed the horizontal tail to provide trim during the changeover from aerodynamic lift to powered lift (supplied by the lift fans). For an unknown reason, the pilot (perhaps inadvertently) actuated this switch at too high an airspeed and because of the out-of-trim condition which resulted, lost control of the aircraft.

Operational considerations for transition also underline the need for flexibility. Early experience with V/STOL aircraft in operational techniques is discussed in reference 59. In this report, the effect of the V/STOL concept on transition flexibility is emphasized. In general, V/STOL concepts which have separate (nonintegrated) power and aerodynamic lift, such as the P-1127, generally have a wider transition corridor compared to a tilt-wing type, and therefore more flexibility for effecting a transition.

5.4 TOLERANCE IN CONVERSION

The need for sufficient leeway in changing from hover to conventional flight with minimum requirements for large attitude changes, precise power programming, etc., is obvious in order to reduce pilot workload. The particular V/STOL concept has a direct effect on this point. For example, in transitioning, the XV-5A fan-in-wing concept (described in ref. 9) transfer from powered lift to aerodynamic lift and vice versa is made by abruptly changing the angle of attack through approximately 12 deg when the exhaust gases are diverted from the wing fan system to the conventional tailpipes for cruise operation. Another example of a V/STOL concept where lack of tolerance in conversion was present is the VZ-3 aircraft. This behavior discussed in the preceding section (5.3) showed the problems which can arise when precise programming of flap position and power are required. Operation of tilt-duct aircraft such as the VZ-4 (described in ref. 60) and the X-22 (ref. 19) showed acceptable characteristics, in that pitch trim changes and programming of duct angles were easily handled by the pilot. As noted in reference 19, however, the pitch and roll dynamics of the X-22A during conversions and reconversions were unsatisfactory. A PIO of 1 to 1.5 sec period was present with peak-to-peak variations of 4 to 6 deg in pitch and roll, which increased pilot workload. The jet-lift types of aircraft, such as the P-1127, VJ-101, and DO-31, were more straightforward in conversion. For the DO-31 aircraft, transition procedures as described in reference 47 showed the benefits of a copilot in carrying out the conversion.

5.5 CONTROL MARGINS

The margins of control remaining about any axis during transition is difficult to specify in quantitative terms because of the lack of data and the many interrelated factors which must be considered. Margins are dictated primarily by the amount of maneuvering required and the sensitivity of a particular V/STOL

concept to gust disturbances. To provide an arbitrary value of 50 percent remaining, as was done in AGARD Report No. 408A, may unduly penalize some V/STOL concepts. For example, operation of the DO-31 aircraft in transition, discussed in reference 47, indicates that nearly full nose-down pitch control was needed in the speed range of 60 to 80 knots to counteract aerodynamic and power-induced moments. In all of the operational flying of the aircraft, the lack of a control margin in pitch was not considered to be a problem because the aircraft was not sensitive to gusts in this speed regime and maneuvering in a nose-down direction was not required. In fact, the pilot was not aware of the small margin available in pitch because the attitude command control system employed masked any feedback of trim requirements to the cockpit control stick. In this respect the system has been criticized, since the pilots felt that an indication of control limits is desirable for situation information purposes.

The foregoing example was not meant to imply that a margin in control about all axes is not mandatory, but rather to point out that fixed values such as a 50 percent margin may not be meaningful. Clearly, the need for some amount of control margin has been illustrated by operational experience with several jet-lift aircraft such as the P-1127, BALZAC, and Mirage III-V, where lack of control margin in roll during transitions resulted in loss of the aircraft. In these cases where the rolling moment due to sideslip is powerful enough to dominate the response of the aircraft, a 50 percent margin may not be adequate to prevent severe upsets, and other means may be required to circumvent the problem. In fact, this precaution has been taken on the Harrier aircraft (see ref. 29) where, as previously noted, a shaking device (warning system) has been attached to the rudder pedals to help limit sideslip excursions.

5.6 TRIM CHANGE

The need to limit maximum forces about all axes during transition is in keeping with good operational practice. The magnitude of forces allowed will depend in part upon the duration over which they must be applied. The forces are small on the assumption that they will not be trimmed out. The forces presented in the Criteria 5.6 of Part I are similar to those previously discussed elsewhere in Part II and reflect pilots' comments from operational experience. For example, the X-22A tilt-duct aircraft (ref. 19) required slightly less than 30 sec in transition, and the pitch force changes varied from 2 to 6 lb in accelerating and decelerating transitions, respectively. Pilots' comments were that significant longitudinal trim changes were required in reconversions, and that lateral and directional trim changes were significant only in a crosswind transition. A further examination showed that the pilot disliked the change in the direction of the forces (and control displacement) which occurred partway through the transition. For the XC-142A aircraft (ref. 5), the tests showed that in making a fuselage level transition, negligible pitch and roll forces were required; however, rudder pedal forces were too large (PR 4). For the CL-84 (ref. 6), the lateral and directional trim changes were negligible in either direction. Pitch force changes were of the order of 3 lb.

5.7 RATE OF CONTROL MOVEMENT

The intent of the criteria on rate of control movement during transition is to limit the rapidity of divergence. Although a value is specified only for the pitch control, corresponding requirements exist for roll and yaw; however, quantitative data on allowable values are not available. In fact, the values for the pitch axis are somewhat arbitrary because the tolerable amount depends on several interrelated factors, including the rapidity desired for completing the transition, the direction of the trim change, and the magnitude of forces required, which, in turn, depend upon the control gradient. The value of 1 in./sec came from very early studies using a variable stability helicopter and does not necessarily reflect operational experience.

Section 6.0

MISCELLANEOUS CHARACTERISTICS

6.1 GENERAL

Documentation of the criteria for this section, which deals with handling qualities items that do not fall into the traditional stability and control sections, is chiefly qualitative because of the nature of the material.

6.2 GROUND HANDLING - GENERAL

The backup material for ground handling was obtained primarily from pilots' comments of the behavior of various V/STOL airdraft during routine testing and operational trials. For the most part, it is not possible to set quantitative limits on these items because of the nature of the material and the fact that systematic tests have not been conducted for this purpose.

6.2.1 Landing Gear. The dynamic requirements for the landing gear of V/STOL aircraft are severe since they must be able to cope with all types of surface conditions for both vertical and roll-on landings. The gear must provide the proper shock-absorbing capabilities for uneven surfaces and hard landings and yet be firm enough to indicate when the wheels touch the ground. Ground contact must be apparent to the pilot on V/STOL aircraft for two reasons. First, in landings the relatively high thrust must be reduced as soon as possible to minimize ground erosion and ingestion and, second, rebounding back into the air tends to occur more easily under high-thrust conditions. This behavior was evident in unpublished data of the TriPartite Trials of the P-1127 Kestral and in US Air Force tests (ref. 7). The outriggers had an adverse effect on landing dynamics such that, if the main gear did not touch down first, marked lateral-directional disturbances occurred. Too high a sink rate could result in the aircraft bouncing back into the air in an unfavorable attitude with the possibility of incurring structural damage on the second ground contact (8 ft/sec design limit). In general, soft landings were difficult to achieve because of unsteady behavior of the aircraft due to ground effect disturbances (recirculation) and the uncertainty that touchdown had occurred. This problem was not peculiar to jet-lift types, as a similar problem occurred on the CL-84 (ref. 6). In this case, a positive ground effect tended to arrest a nominal descent rate and the pilot had to reduce power very slowly to "hunt" for the ground. The soft stroke landing gear made it difficult to judge when the aircraft was actually on the ground, and the pilot tended not to reduce power which prolonged the ground effect disturbances and added to the illusion of being airborne. A green "touchdown" light, activated by oleo compression, was installed to indicate ground contact. A narrow gear track such as used on the XV-5A aircraft (ref. 9) can aggravate the problem because of the tendency to tip over with only small values of lateral motion. The narrow gear and tendency for an oleo strut to stick on the X-19 VTOL aircraft (ref. 10) caused an objectionable leaning tendency even during no crosswind operation. Because the propeller radial lift forces were still effective, the aircraft remained light on the gear down to about 30 knots, resulting in skidding and rocking motions.

For STOL operation, the dynamics of the landing gear are equally important because of performance considerations. In operation of the Breguet 941 aircraft (ref. 52), greater accuracy in touching down on a specific spot could be obtained by using a partial flare technique in which ground contact was made intentionally with a finite sink velocity. This required a soft long-stroke landing gear to reduce structural loads and special nonlinear response characteristics to avoid rebounding back into the air. The dynamic characteristics of this type gear were troublesome, however, in cross-wind operation because it was difficult to maintain a wings-level attitude during rollout. In one landing incident with this aircraft, the propeller and engine pod struck the ground inadvertently during a crosswind landing. In this case, the ground reaction moments supplied by the narrow tread landing gear and the long-stroke struts were not large enough for the rolling moments that occurred during crosswind operation.

6.2.2 Control Effectiveness During Takeoff, Landing Rollout, and Taxi. Experience with various V/STOL aircraft has indicated that control on the ground can be a problem because of reduced cornering forces from the gear and induced flow effects due to engine power. Crosswind can have serious effects on control of ground path during takeoff and landing, depending on the V/STOL concept. Important aerodynamic parameters include side force, directional stability, and dihedral effect. Aircraft with large side-force characteristics, such as the X-22A (ref. 19), are more difficult to hold straight because of the tendency to lean downwind. This tendency was also present on the P-1127 aircraft (ref. 7) which leaned on the downwind outrigger gear with a 4° to 5° tilt angle, primarily causing pilot discomfort with no adverse effect on directional control. On the tilt wing XC-142 (ref. 5), the degree of wing tilt affected ground controllability. In the range of tilt angles from $40^\circ/60^\circ$ to $80^\circ/0^\circ$, the aircraft experienced recirculation of the slipstream, resulting in large lateral-directional disturbances that could not be satisfactorily damped by full deflection of the controls. Another tilt-wing type, the CL-84 (ref. 6), has not at this time experienced this effect. The deflected slipstream Breguet 941 also experienced difficulty in landings in crosswind as noted in the discussion on landing gear (section 6.2.1). During landing rollout, lateral control effectiveness depends on slipstream velocity and when reverse thrust was used to reduce ground roll, insufficient control existed when reverse thrust was used to reduce ground roll.

6.2.3 Power Checks Prior to Takeoff. Operation of several VTOL aircraft has indicated that problems have been encountered during checkout for functioning of controls, engine characteristics, transmission, etc., prior to takeoff. Because the thrust vector angle can be positioned on all VTOL aircraft (except helicopters) to produce large horizontal forces when the aircraft is on the ground, the danger of gear skidding is imminent when high-engine powers are used. In fact, on one occasion with the P-1127, a rolling takeoff (RTO) was attempted with the parking brakes inadvertently left on, resulting in heavy damage to the aircraft because the aircraft could not be controlled directionally. In another case, the XC-142 aircraft was extensively damaged when it jumped the wheel chocks and ran into a hangar at high engine power during engine runup.

6.3 CROSS-COUPLING EFFECTS - GENERAL

The various cross-coupling characteristics are grouped together for discussion pruposes.

6.3.1 Gyroscopic Effects.

6.3.2 Inertial Cross-Coupling Effects.

6.3.3 Mechanical Cross-Coupling. Cross-coupling effects due to gyroscopic, inertial, or mechanical inputs have been encountered in various degrees on VTOL aircraft. The primary effect is a tendency to reduce precision of control by increasing pilot workload. Further, gyroscopic and inertial cross-coupling can be insidious and surprise the pilot because of the development of large unintentional divergences in flight path. For example, during early operation of the Bell X-14 aircraft, pitch-up occurred uncontrollably (in a zero radius loop) on several occasions at altitude because the pilot was not aware of a slight yawing condition. Mechanical cross-coupling effects were quite noticeable on the tilt-prop X-19 aircraft (ref. 10) because of unequal, incremental thrust differences of the various propellers about axes other than the one desired for control.

Systematic studies of gyroscopic coupling effects obtained on a piloted moving-cab simulator (ref. 61) showed a rapid deterioration of pilot rating with increase in gyroscopic coupling. This was true without considering the further adverse effects of gusts. For these reasons, the criteria are intended to completely discourage any amount of gyroscopic coupling.

6.4 MINIMUM FLIGHT SPEEDS - GENERAL

A documentation of this area is limited because of the difficulty of generalizing on the results of specific V/STOL aircraft and the background of material available for conventional aircraft.

6.4.1 Loss of Lift. The definition of what constitutes satisfactory aircraft dynamic response following loss of aerodynamic lift has traditionally been a problem for conventional aircraft, and attempts have been made to quantify the allowable attitude changes in the Mil Spec 8785 (ref. 13). Large lateral attitude changes (rolloff) at the stall are particularly undesirable because of the excessive loss of altitude resulting during the recovery process. For most V/STOL aircraft, the dynamic behavior at the stall has been more gentle because powered lift is shared with aerodynamic lift to support the aircraft in forward flight. In addition, power effects have a marked influence on stalling behavior, depending on the V/STOL concept. When the thrust is vectored downward through large angles such as on the tilt-wing XC-142, for example, reducing power increases the intensity of buffet and increases the amount of settling. On the other hand, the application of engine thrust has an immediate beneficial effect of unstalling the separated flow on the wing by reducing angle of attack and reenergizing the boundary layer. As a consequence, these aircraft types can make a landing approach in a partially "stalled" condition without seriously affecting flight-path control. For aircraft where there is less interaction between the aerodynamic lift surfaces and power, such as the P-1127, the consequences of stall are usually more severe and a larger margin in angle of attack from the stall must be maintained to avoid a pitchup region existing at higher α.

No attempt was made in the criteria to quantify the allowable attitude divergence following loss of aerodynamic lift for several reasons. Foremost is the difficulty of flight testing for the stall condition. For reasons discussed previously, most V/STOL aircraft do not exhibit a clearly defined stall but rather a gradual increase in buffet intensity, accompanied by a progressive increase in rate of descent. Further, it is unrealistic to assume a fixed power condition in the stall (as is done for conventional aircraft) because power adjustment has assumed a primary role for control of the aircraft in STOL operation. These criteria may be revised after more operational experience has been obtained to set allowable margins of benavior.

6.4.2 Warning of Approach to V_{min}. The definition of an adequate amount of warning to approach to V_{min} is another difficult area to define quantitatively for V/STOL aircraft. The purpose of providing stall warning is to make it possible for the pilot to avoid reaching V_{min} unintentionally, thereby providing a margin in angle of attack to use for changing the flight path (maneuvering) and for gusty air operation. For aircraft such as the tilt-wing XC-142 (ref. 6), prestall buffet occurs over too large a speed range to serve as an effective margin. For other propeller STOL types typified by the Brequet 941 (ref. 52), a margin of 5 knots or 5° angle of attack was found to be adequate as a minimum to allow the pilot to take corrective action. Additional work needs to be done to relate the amount of warning to the severity of the stall. For aircraft that exhibit only mild buffeting and small flight-path divergence at the stall, less stall warning margin is needed.

6.5 WARNING OF APPROACH TO HAZARDOUS FLIGHT CONDITIONS

Several V/STOL aircraft have been flown into hazardous flight regions unintentionally with serious consequences. For example, a hazardous area for several jet-lift VTOL aircraft is that associated with sideward flight (side-slipping) at low forward speeds, where large rolling moments were produced, particularly at high values of angle of attack. In these cases, because of the suddenness with which the rolling moment developed, corrective action could not be taken in time to prevent loss of control and the aircraft were destroyed. In fact, because the instinctive reaction of the pilot was to try to raise the low wing with lateral control, the fundamental cause of the rolling moment (sideslip) was not reduced and, as a consequence, large bank angles of over 90° were attained, with the resultant large loss in altitude. Unpublished wind-tunnel tests showed that the cause of the large (and sudden) increased rolling moment due to sideslip was associated with the induced flow from the vertical lift engine(s). Other VTOL types, such as the tilt-wing XC-142, also experienced problems due to sudden pitchup during deceleration as well as unsteady flight due to recirculation in ground effect.

Because these aircraft can be flown into hazardous areas quite inadvertently, and the causes of the undesirable aircraft behavior may be fundamental to the lift concept, warning or other devices are necessary to restrict operation into these parts of the flight envelope.

It is not possible to quantify these criteria at this time; however, a precautionary note is made in reference 3 to ensure that desirable parts of the flight envelope are not needlessly restricted.

6.6 AIRCRAFT BEHAVIOR FOLLOWING SYSTEM FAILURE

Most V/STOL aircraft will have complex systems required for low-speed operation. Failure of a stability augmentation system, for example, can result in an intolerably high pilot workload, particularly if the pilot must adapt to a new set of gains quickly. Further, a partial loss of SAS may not be apparent to the pilot because the effect may be only to reduce authority limits. For this case, problems may be experienced quite suddenly when large-magnitude excursions are commanded or when encountering turbulence. Other failures such as with BLC systems or lift engines may appear insidiously as an altitude loss without any accompanying angular motions, particularly when using attitude command control systems.

It is not possible to specify allowable attitude limits for aircraft behavior following systems failures because of the lack of systematic test results to apply to this area. In addition, there are many interrelated effects that make it difficult to establish a generalized set of numbers that would avoid unduly penalizing a particular design.

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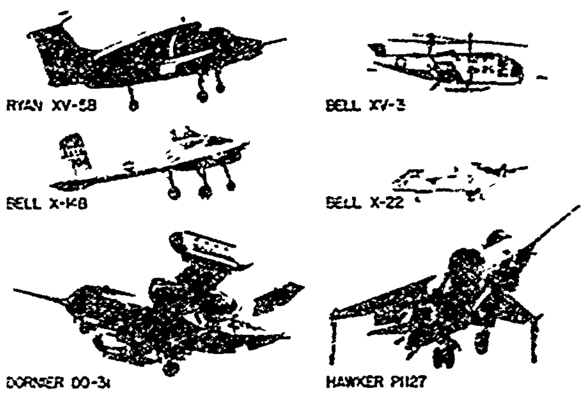

Figure 1. Composite Photograph of VTOL Aircraft.

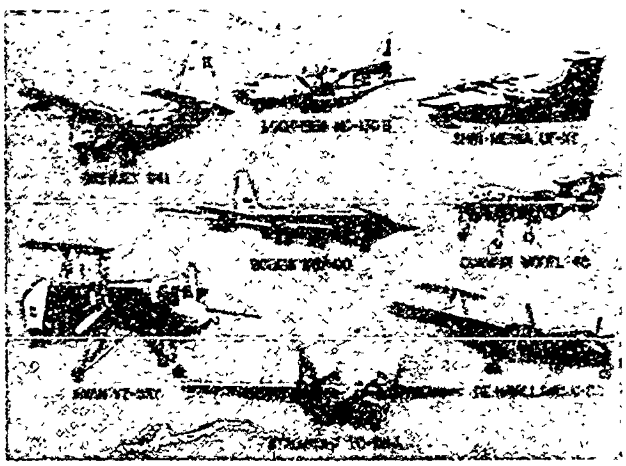

Figure 2. Composite Photograph of STOL Aircraft.

Aircraft	Pitch control system: Control travel (in.)	Force gradient (lb/in.)	Breakout force (lb)	Friction force (lb)	Comments
VZ-4	±6.0				
VZ-2	±5.2				
X-14	±6.0	0		0.5	Last inch of control travel was not effective. No comment on zero force gradient
SC-1	±3.5	2.0	2.0	1.0	
SG-1262	±4.0	3.0	0.5	2.0	(VAK 191 hover rig) only first 2.0 to 2.5 in. of pitch control is effective
XV-5A	6.0 aft 6.5 fwd	0 to 0.95	0.7 to 3.6	2.0 to 3.0	Breakout force and force gradient varied as a function of vector angle. (Values given for first in. of travel)
CL-84	4.2 aft 2.5 fwd	2.0	0.5		Hover force gradients satisfactory. High gradient desired to help stabilized pitch attitude at intermediate wing angles
P1127	7.5 aft 3.75 fwd	1.2	1.5	1.0 to 6.0	In hover, only first 1.2 in. of pitch control travel is effective
VJ-101-X1	±6.0	1.1	1.9		
X-22	±5.6	1.5	0.5 to 1.0 0.6 to 0.8 1.75 to 5.0	Low	Duct angle = 0°. Trim and feel system on. Duct angle = 90° Trim and feel system on. Duct angle = 90° Trim and feel system off
XC-142	6.2 aft	1.75	2.5		60° wing angle. only first 1.2 in. of forward and 5.2 in. aft control travel were effective
XC-142	5.2 aft 4.2 fwd	4.2	1.0		0° wing angle. last 2.0 in. fwd travel was free play.
DO-31	±5.2	3.2	1.54		
VZ-3RY	7.0 aft 7.0 fwd				
CV-48	2.9 aft 4.1 fwd				
UF-XS	8.2 aft 4.0 fwd	2.5	6.0	5.0	Pilot rating = 3
C-8A	7.6 aft 4.7 fwd				
BR941	5.0 to 6.0 aft 4.5 fwd	4.0 (1963 data) 5.0 (1966 data)	2.0 (1963 data) 3.0 (1966 data)	10.0 (1963 data) 5.0 (1966 data)	Pilot rating satisfactory
YC-134A	9.0 aft 5.0 fwd	1.0	15.0	30.0	Objectionable high friction and high inertia.
NC-130B	9.5 aft 7.0 fwd	7.5	6.0	6.0	Pilot rating undesirable.
367-80	10.2 aft 5.0 fwd	3.5	6.0	6.0	

Figure 3. Pitch Control System Mechanical Characteristics.

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	Roll control system				
Aircraft	Control travel (in.)	Force gradient (lb/in.)	Breakout force (lb)	Friction force (lb)	Comments
VZ-4	±6.0				
VZ-2	±4.2				
X-14	±5.0	0		2.0	No comment on zero force gradient – gradient added later so pilot could evaluate attitude system.
SC-1	±3.5	1.1	2.0	1.0	
SG-1262	±4.0	3.0	2.0	2.0	Only first inch of control travel was effective. Pilot rating satisfactory except when pilots used more control than needed to get maximum angle, then complained of high forces.
XV-5A	3.9 rt 3.2 lt	0.4 to 1.4	0.7 to 2.0	1.0 to 1.8	2.0 pilot rating – forces too high for low speed maneuvering.
CL-84	±5.2	2.0	0.5		Satisfactory pilot rating.
P1127	±4.25	1.5	1.0	1.5	Satisfactory pilot rating.
VJ-101-X1	±5.5	0.73	1.9		
X-22	±5.2	1.5	0.4 to 0.8 0.5 to 0.6 4.25 to 4.5	Low	0° duct angle – trim and feel system on. 50° duct angle – trim and feel system on. 90° duct angle – trim and feel system off.
XC-142	±3.0	1.5	0.1 to 2.5		
DO-31	±4.65	3.0	1.76		
VZ-3RY	±7.5				Deflect too large, stick hits pilot's knee.
CV-48	±4.0				Deficient due to high mass and inertia.
UF-XS	±14.0 (±105°) wheel control	1.0	6.0	10.0	Pilot rating = 3.5 – wheel throw too large, desire 60°.
C-8A	±9.0 (±80°) wheel control	1.0	3.0	6.0	
BR941	±4.5	3.0 (1963 data) 4.0 (1966 data)	2 (1963 data) 3 (1966 data)	4.0 (1963 data) 3.0 (1966 data)	Satisfactory pilot rating. Satisfactory pilot rating, but lag in propeller pitch.
YC-134A	±16.0 (±120°) wheel control	1.0	15.0	30.0	Objectionable high inertia.
HC-130B	±11.7 (±90°) wheel control	1.0	8.0	15.0	6.5 pilot rating.
367-80	±8.3 (±75°) wheel control	1.3 @ 90 knots 1.5 @ 115 knots	6 @ 90 knots 4 @ 115 knots	2.0	Satisfactory pilot rating. 3.5 pilot rating.
367-80	±8.3 (±75°) wheel control	5.5 @ 115 knots	4 @ 115 knots	2.0	4.5 pilot rating.

Figure 4. Roll Control System Mechanical Characteristics.

	Yaw control system				
Aircraft	Control travel (in.)	Force gradient (lb/in.)	Breakout force (lb)	Friction force (lb)	Comments
VZ-4	±3.5				
VZ-2	±3.0				
X-14	±3.0				
SC-1	±3.0	8.0		5.0	
SG-1262	±4.0	13.0	4.0		75% of yaw control power obtained in first 0.75 in. of travel. 25% in last 3.25 in.
XV-5A	±3.5	0 to 4.7	1.0 to 5.0	4.0	Lower breakout and gradient forces produced a "no feel" system in hover.
CL-84	±3.2	7.0	3.0		
P1127	±2.25	12.0		7.0	
VJ-101-X1	±3.15	30.3	4.4		High pedal forces used to ease dynamic structural problems of the heavy swivelling engine pods used for yaw control. Peak force 130.
X-22	±3.25		8.0		
XC-142	±2.5	60-75	2 to 10		Directional force gradient too high. Should be reduced 50%. Poor harmony with other axis.
DO-31	±2.05	30.2	8.8		
VZ-3RY					
CV-48	±3.25				
UF-XS	±2.5	20.0	3.0	5.0	Rudder force gradient too low, desire 50% more.
C-8A	±3.6				
BR941	±2.8	30 (1963 data) 34 (1966 data)	10.0	3.0	Satisfactory pilot rating.
YC-134A	±4.25	3.0	22.0		Objectionable.
HC-130B	±3.5	30.0			6.5 pilot rating – bad hysteresis.
367-80	±2.5				

Figure 5. Yaw Control System Mechanical Characteristics.

① LIMIT FOR MAXIMUM BREAKOUT FORCE AND MAXIMUM FORCE GRADIENT
② LIMIT FOR MINIMUM BREAKOUT FORCE AND MINIMUM FORCE GRADIENT
③ LIMIT FORCE FOR FULL CONTROL DISPLACEMENT FROM TRIM
④ MINIMUM AND MAXIMUM BREAKOUT FORCE LIMITS
⑤ MINIMUM CONTROL TRAVEL = ±4.0 in.
⑥ MAXIMUM CONTROL TRAVEL = ±6.5 in.

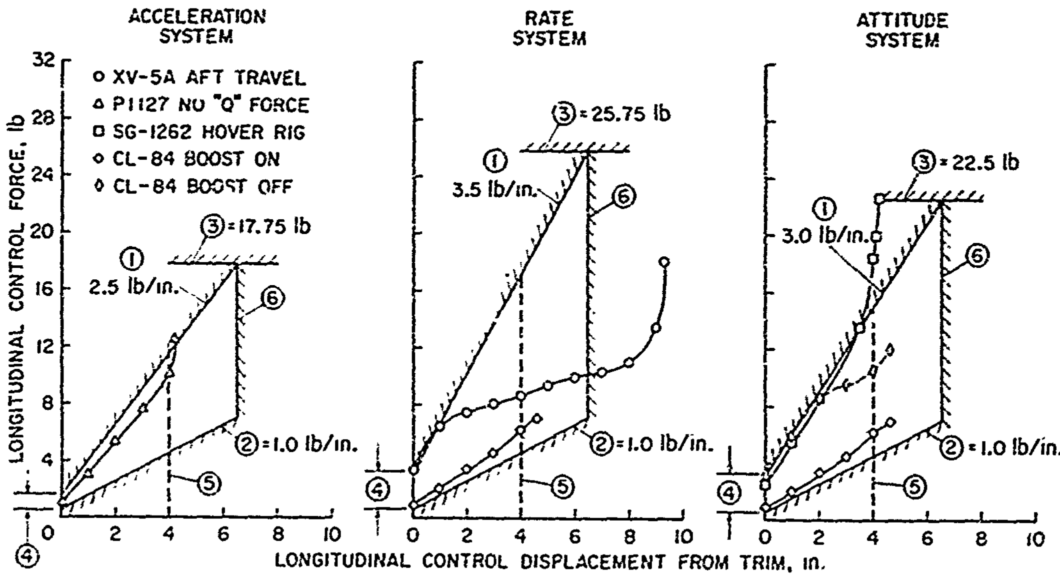


Figure 6. Longitudinal Control Force Documentation; Hover and Low Speed Flight Condition.

① LIMIT FOR MAXIMUM BREAKOUT FORCE AND MAXIMUM FORCE GRADIENT
② LIMIT FOR MINIMUM BREAKOUT FORCE AND MINIMUM FORCE GRADIENT
③ LIMIT FORCE FOR FULL CONTROL DISPLACEMENT FROM TRIM
④ MINIMUM AND MAXIMUM BREAKOUT FORCE LIMITS
⑤ MINIMUM CONTROL TRAVEL = ±3.0 in.
⑥ MAXIMUM CONTROL TRAVEL = ±6.5 in.

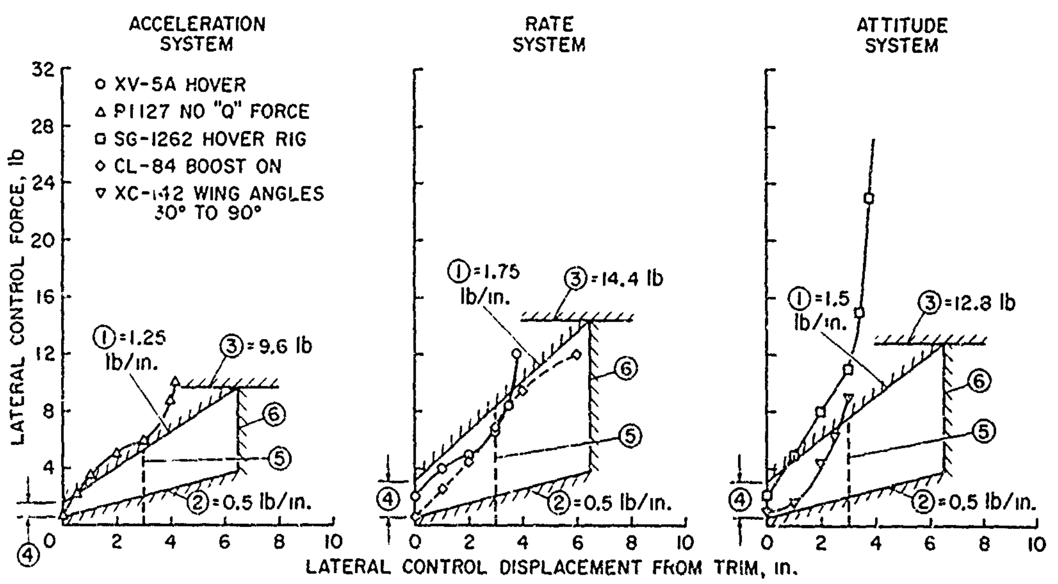


Figure 7. Lateral Control Force Documentation; Hover and Low Speed Flight Condition.

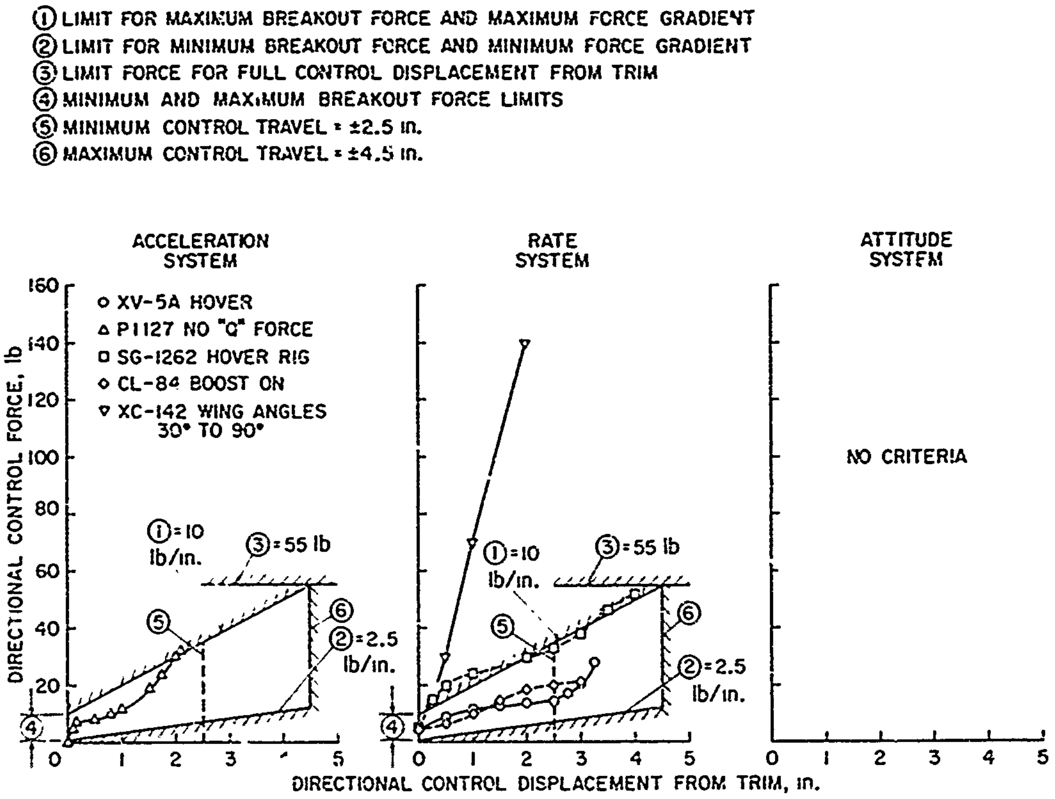


Figure 8. Directional Control Force Documentation; Hover and Low Speed Flight Condition.

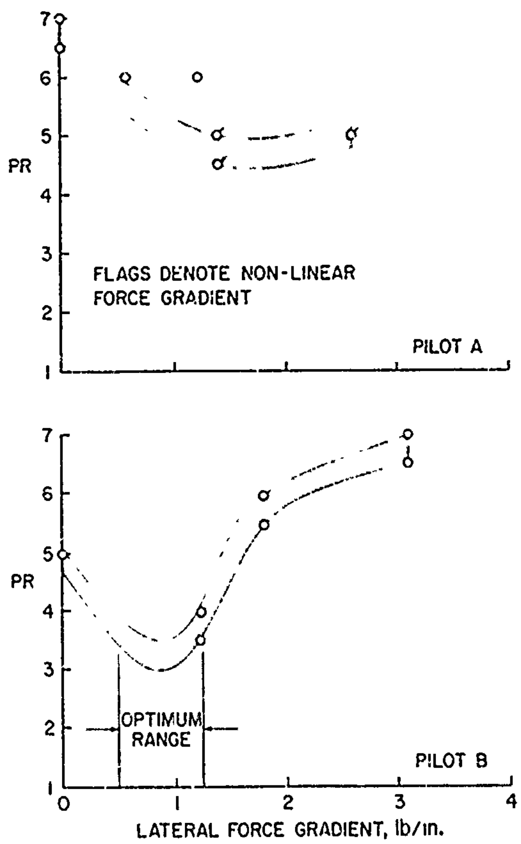


Figure 9. Control Force Characteristics Study.- Acceleration System.

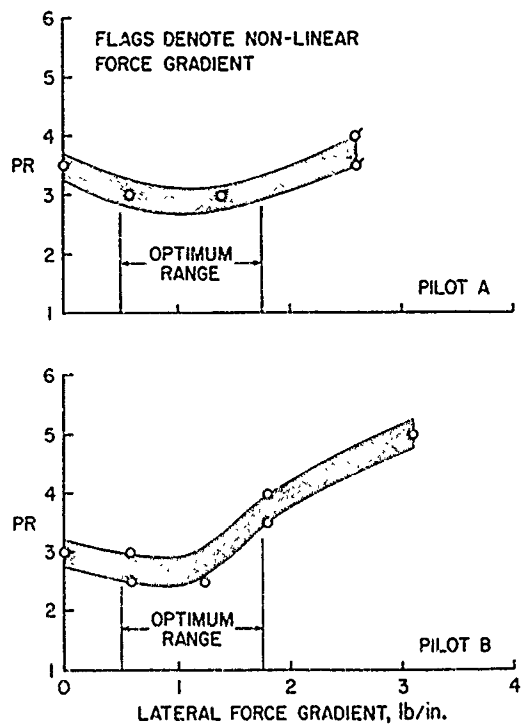


Figure 10. Control Force Characteristics Study - Rate System.

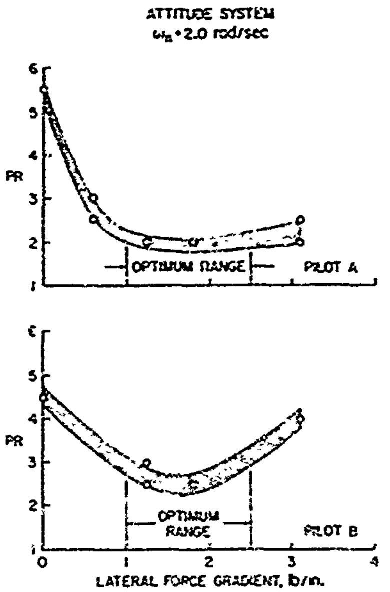


Figure 11. Control Force Characteristics Study - Attitude System.

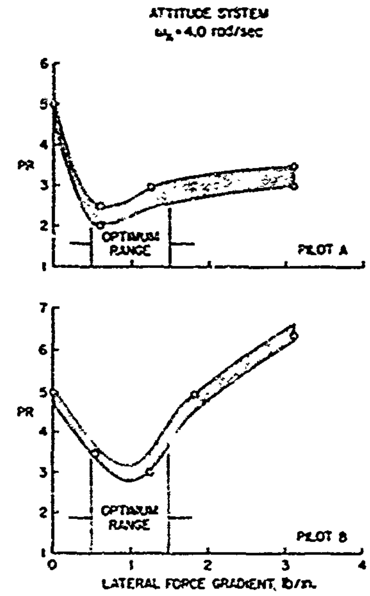


Figure 12. Control Force Characteristics Study - Attitude System.

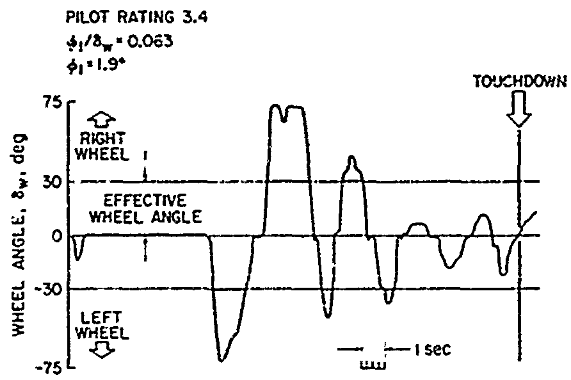


Figure 13. Time History of Wheel Motion During Sidestep Maneuver.

Aircraft	Gross weight (lbs)	I_{yy} moment of inertia (lb-sec²-ft)	Pitch control power (rad/sec²)	Pitch control sensitivity (rad/sec²/in.)	Inherent rate damping (1/sec)	Total rate damping (1/sec)	Flight condition	Comments
VZ-4 tilt-duct	3 200	1,500	±0 6	±0 1			Landing approach	100% control power used for trim during landing approach
VZ-2 tilt-wing	3,730	3,450	±0 5	±0.09	-0 15	-0.15	Transition at 30 knots	Full span flap reduced trim requirements from 30 to 10%.
X-14 Jet-lift	3,500	3,400	±0 44	±0 08	-0 15		Hover	100% control power used for trim when thrust vectoring from hover Pilot rating for sensitivity & damping = 5-1/2 - 6
X-14 Jet-lift	3,900	3,400	±0 8	±0 16	-0 15		Hover	Pilot rating for sensitivity and damping = 2-1/2.
SC-1 jet-lift	6,900	5,480	±1.1	±0.31	-0 8	-3 0 to -9.8 -3 5 preferred	Hover	High damping requires high control power.
SG-1262 hover rig	8,580	25,900	±0 98	±0 26 (bleed air only)			Hover	Attitude stabilized θ_{max} = ±15°, ζ = 0.6, ω_n = 1.9. Control power = 0.48 bleed air + 0 5 thrust modulation
XV-5A fan-in-wing	9,200	15,139	+1 24 -0 43	±0.10 to ±0 13		-2.5 to -3 2	Hover	Response a function of height control position
XV-5A fan-in-wing	9,200	15,139	+1.00 -0 32	±0 08 to ±0 25		-2 0 to -5.4	Transition	+50% control power used for trim at 43 knots.
CL-84 tilt-wing	10,600	18,700	±1.35	±0 41	-0.4	-4 0	Hover	Attitude stabilized, attitude stiffness (Mθ) = 1.8 sec⁻² High damping requires high control power. θ_{max} = ±13°.
P1127 jet-lift	12,000	22,450	+0 53 -0 97	+0 15 -0 40	-0.8	0 8	Hover	90 - 95% control power used during vertical landing.
P-1127 jet-lift	12,000	22,450	+0 69 -0 43	±0 1			STOL 40 knots	95 - 100% control power used during STOL landing
YJ- 91-XI Jet-lift	13,376	17,540	±1 50			-0.85		Attitude stabilized, ω_n = 3 0 rad/sec², ζ = 0 8, θ_{max} = 16°, θ°/in. of stick = 3.16
X-22 tilt-duct	14,364	32,094	±3 25	±0 45 to ±0 83	-0.5	-5 0 to -7.5	Hover	High control power and damping required for V/STOL research role.
MIRAGE III V Jet-lift	26,240	104,059	±0.39	±0.165		-4.25	Hover	
MIRAGE III V Jet-lift	26,240	104,059	±0.60			4 65	100 knots	
XC-142 tilt-wing	36,900	100,000	±0 45	±0 14			33 - 36 knots 30° wing angle	100% control power was insufficient to trim out propeller pitching moments when accelerated with pitch attitude
DO-31 Jet lift	42,300	206,310	0 42				Hover	ω_n = 2.5, ζ = 1.0, θ_{max} = ±22.0°, θ°/in. of stick = 6 (non linear), 80% of pitch control power required at 40 knots to trim nose-tip pitching moment.

Figure 14. Pitch Control Response Characteristics of VTOL Aircraft.

Aircraft	Gross weight (lbs)	I_{yy} moment of inertia (lb-sec²-ft)	Pitch control power (rad/sec²)	Pitch control sensitivity (rad/sec²/in.)	Pitch rate damping (1/sec)	Time for 10° attitude change (sec)	Attitude change in one sec (deg)[a]	Airspeed (knots)	Comments
VZ-3RY	2,900	2,570	2.0		3.0			60	
CV-48	7,730	6,000	+1.4 -1.0	0.2	0.8	0.8	18 14[b]	55-60	
UF-XS	32,000	173,500	+0.55 -0.21	0.06	1.1	1.2	6.7	55	
C-8A	34,000	215,000[c]						70	
BR941	38,500	140,000	+1.05 -0.75	0.17	1.4	0.9	11.8	60	
BR941	38,500	140,000	+0.30 -0.40	0.10	1.0	1.2	6.6	60	
YC-134A	55,500	225,000	0.40	0.06	1.4	1.5	4.5	70	
C-130B	100,000	900,000	+0.65 -0.45	0.07	1.9	1.2		70	
NC-130B	100,000	900,000	+1.05 -0.75	0.11	1.9	1.0		60	
367-80	150,000	2,250,000	0.24 -0.20	0.03				85-90	
367-80	150,000	2,250,000	+0.43 -0.36					115	

[a]Computed for single degree of freedom with 0.1 sec transport lag and 0.3 sec ramp input

[b]Measured

[c]At 38,000 lbs.

[d]From unpublished data

Figure 15. Pitch Control Response Characteristics of STOL Aircraft.

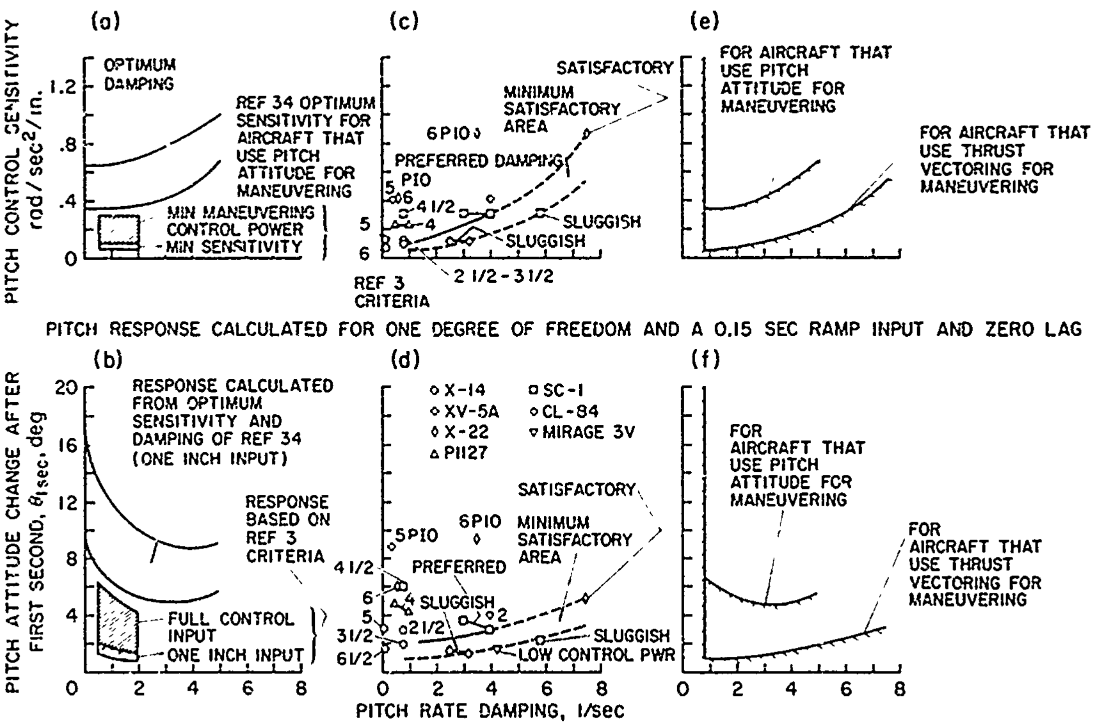


Figure 16. V/STOL Pitch Response Data and Criteria.

Vehicle	Control System Mode	Average Linear Mean Values, Rad/Sec²					
		Stabilization	Pitch Task	Roll-Axis*	Vertical Takeoff	Vertical Landing	Roll-on Landing
Hover Rig	Attitude	0.042	0.065	0.049	0.061	0.056	-
Hover Rig	Rate	0.036	0.062	0.052	0.070	0.065	-
VJ-101-X1	Attitude	0.047	0.075	0.051	0.063	0.067	-
VJ-101-X2	Attitude	0.057	-	-	0.087	0.077	-
VJ-101-X2	Attitude	0.057*	-	-	0.080**	-	0.120**

* Roll Control Power Usage During Pitch Maneuver

** With Afterburners

Figure 17a. Pitch Control Power Usage for Different Tasks (Ref. 14).

Vehicle	Control System Mode	Gross Weight lb	Maximum Values Covering 95% of Total Usage Rad/Sec²			
			Pitch Task	Vertical Takeoff	Vertical Landing	Roll-On Landing
Hover Rig	Attitude	4410 to 5292	0.215	0.26*	0.244	-
VJ-101-X1	Attitude	12680 to 13450	0.246	0.273	0.291	-
VJ-101-X2	Attitude	13010 to 13781	-	0.215	0.281	-
VJ-101-X2	Attitude	14000 to 17310	-	0.230*	-	0.320*
* With Afterburners						

Figure 17b. Maximum Pitch Control Power Usage for Different Tasks (Ref. 14).

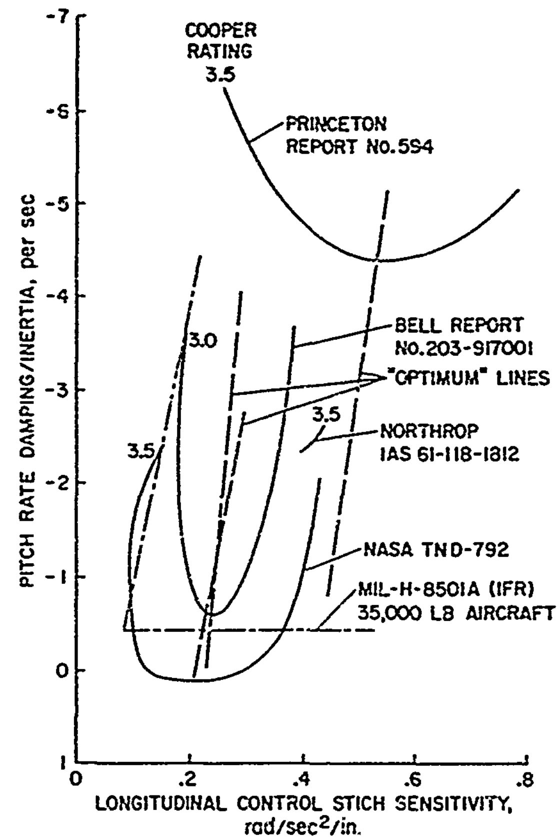


Figure 18. Comparison of Sensitivity and Damping Boundaries.

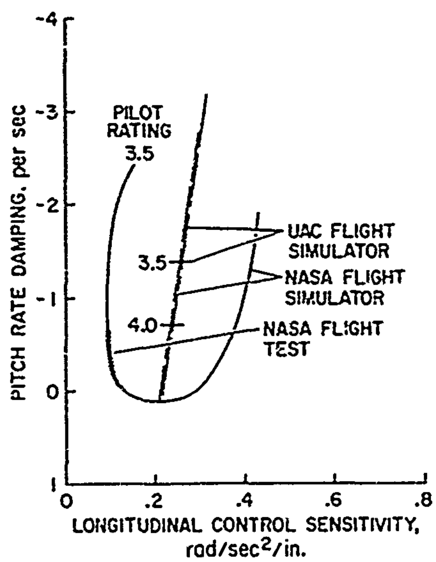


Figure 19. Comparison of Flight and Simulator Data.

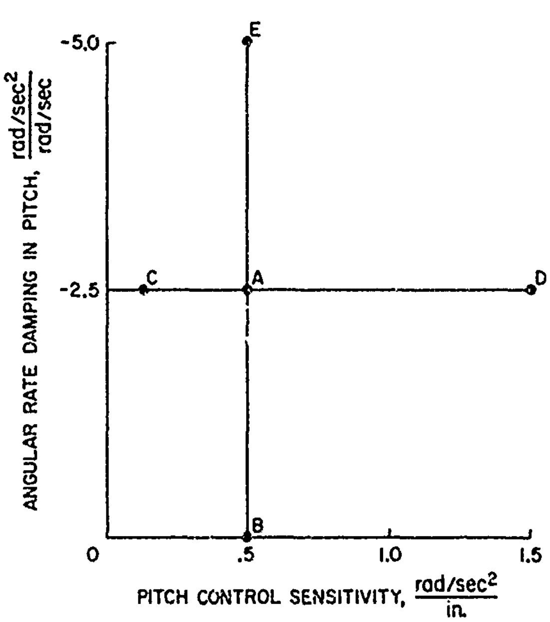


Figure 20. Combinations of Pitch Characteristics Tested.

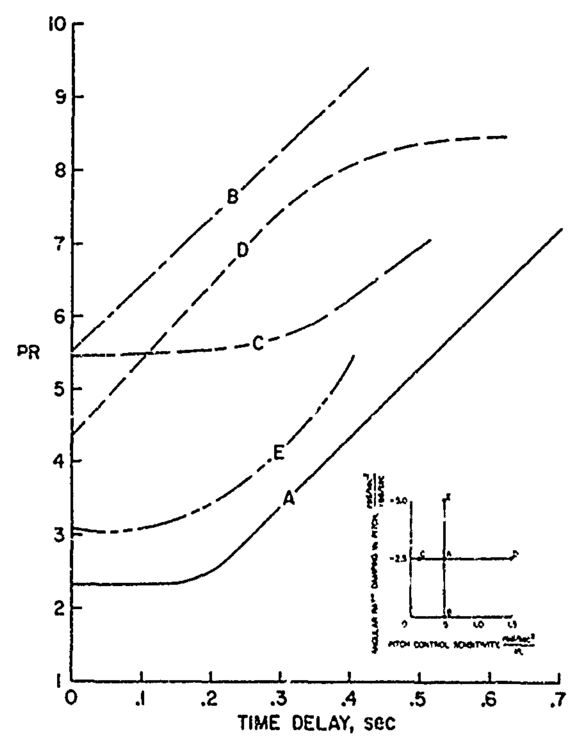


Figure 21. Transport Time Delay Results - Approaches.

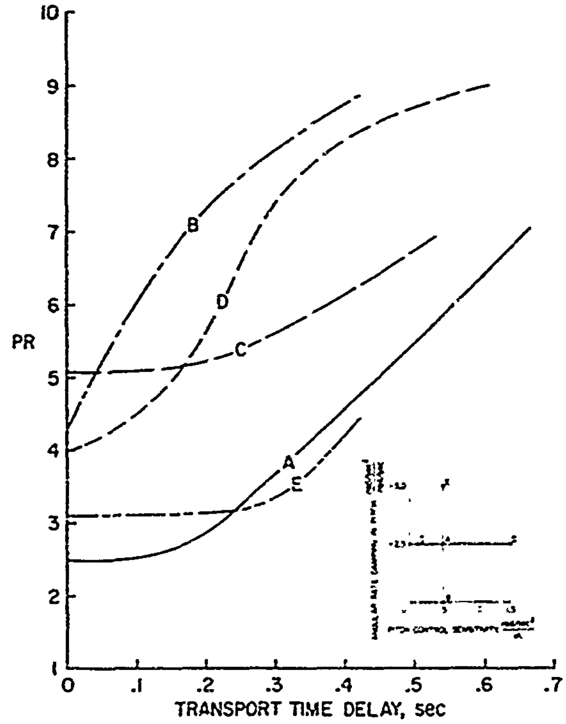


Figure 22. Transport Time Delay Results - Hovering Task.

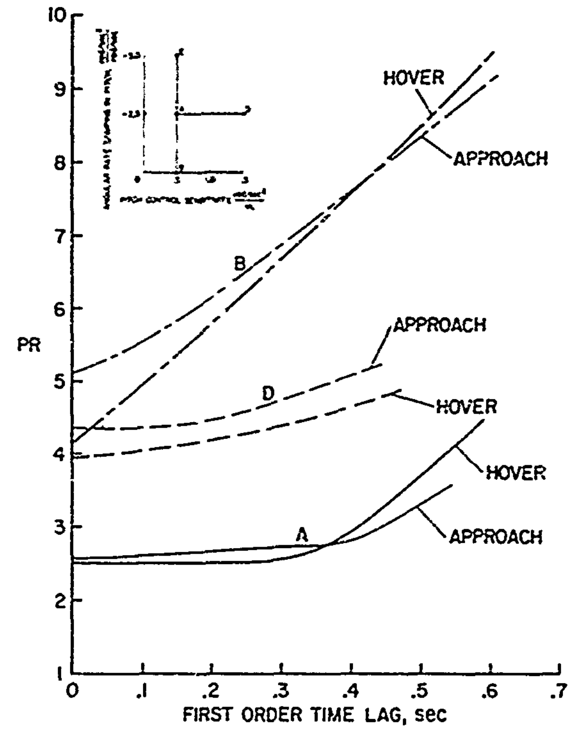


Figure 23. First-Order Time Lag Results.

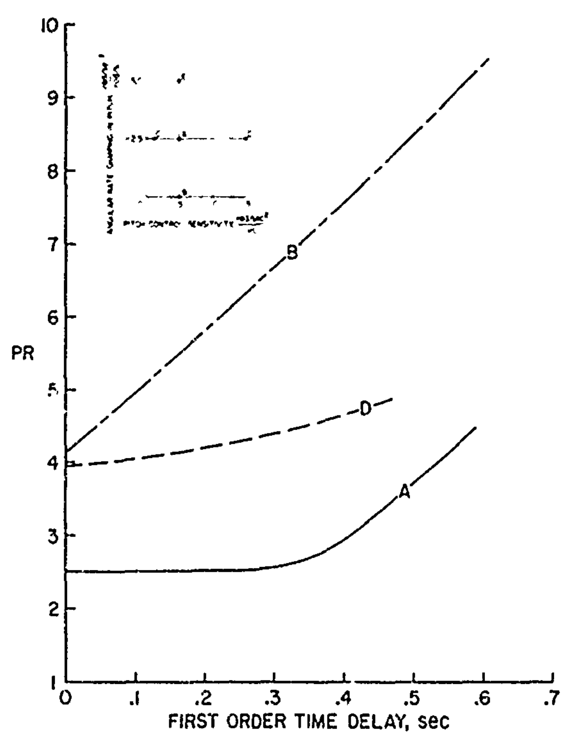


Figure 24. First-Order Time Delay Results - Hovering Task.

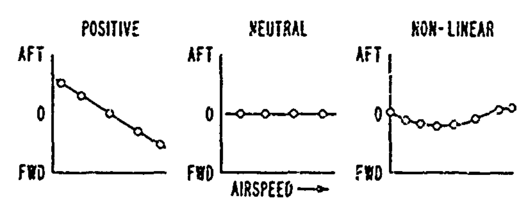


Figure 25. Examples of Trim-Speed Stability.

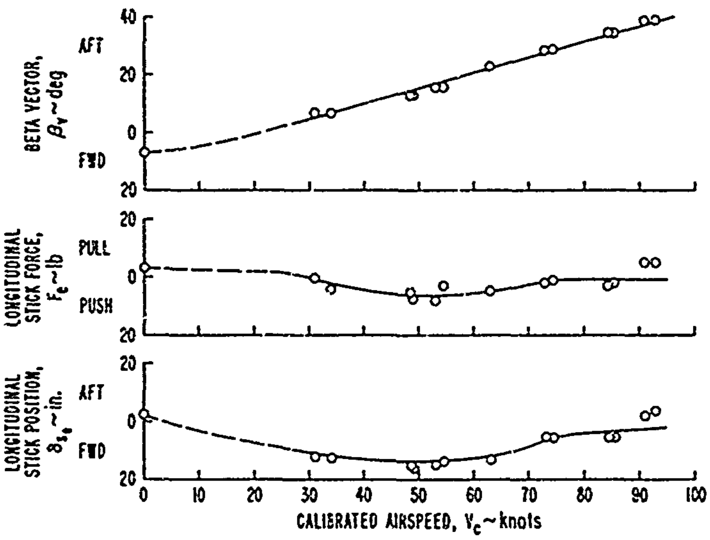


Figure 26. XV-5A Static Longitudinal Trim Stability - Fan Mode (Ref. 9).

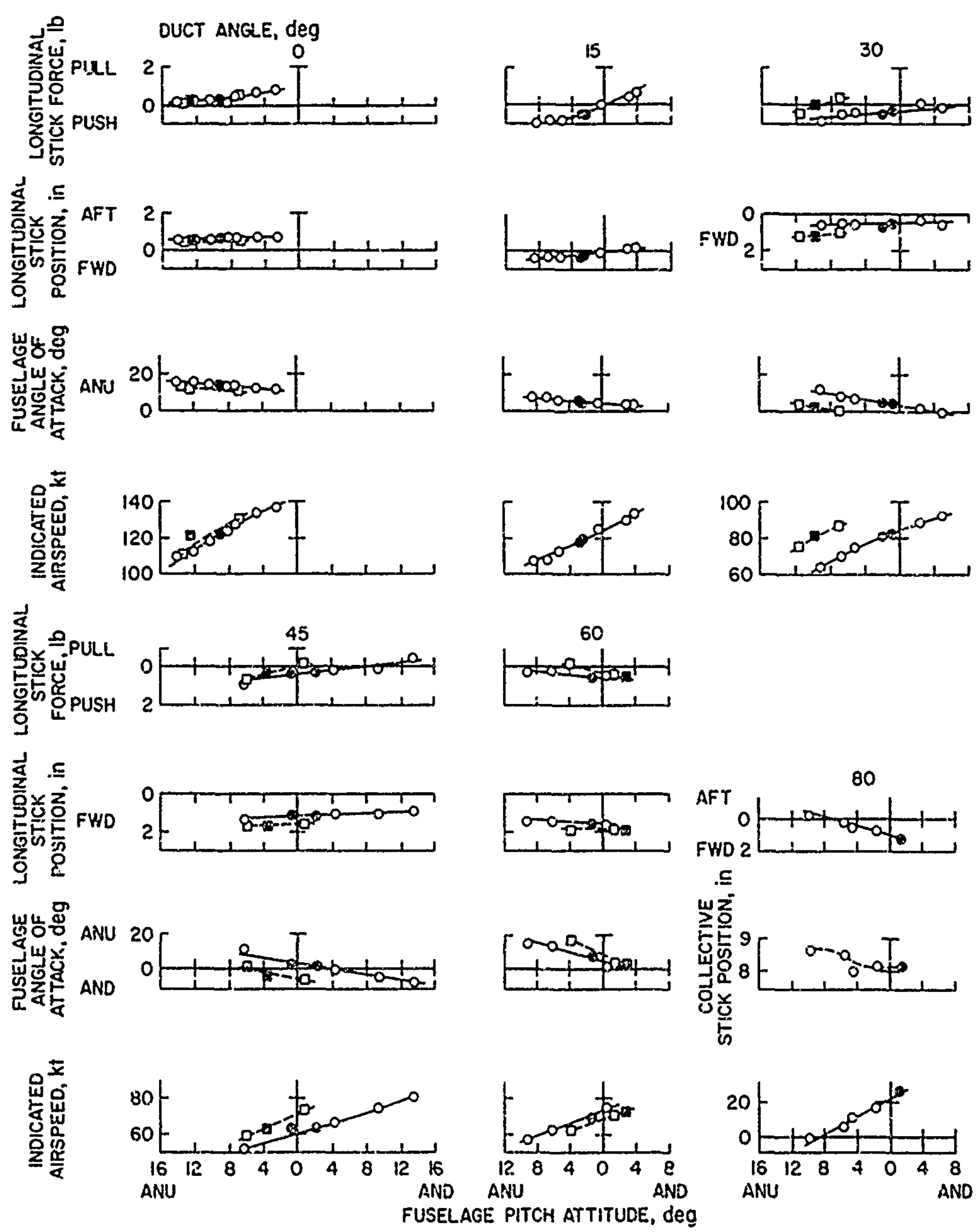


Figure 27. X-22A Static Longitudinal Stability (Ref. 19).

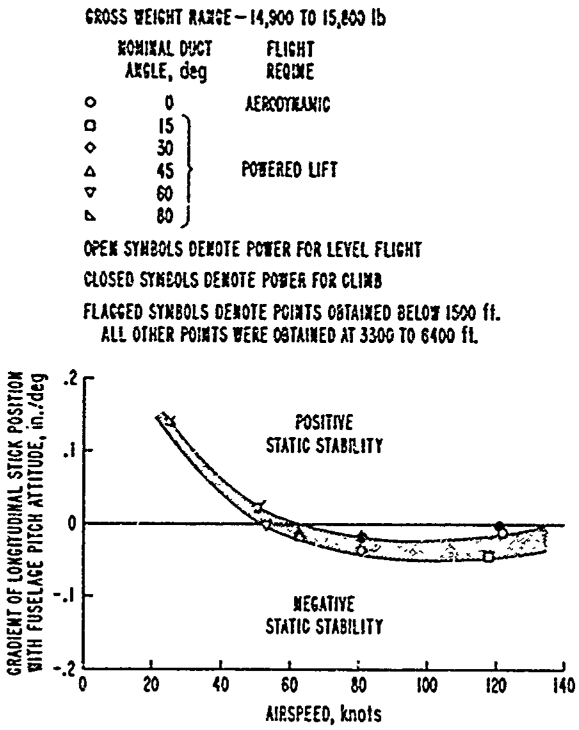


Figure 28. Summary of X-22 Static Stability.

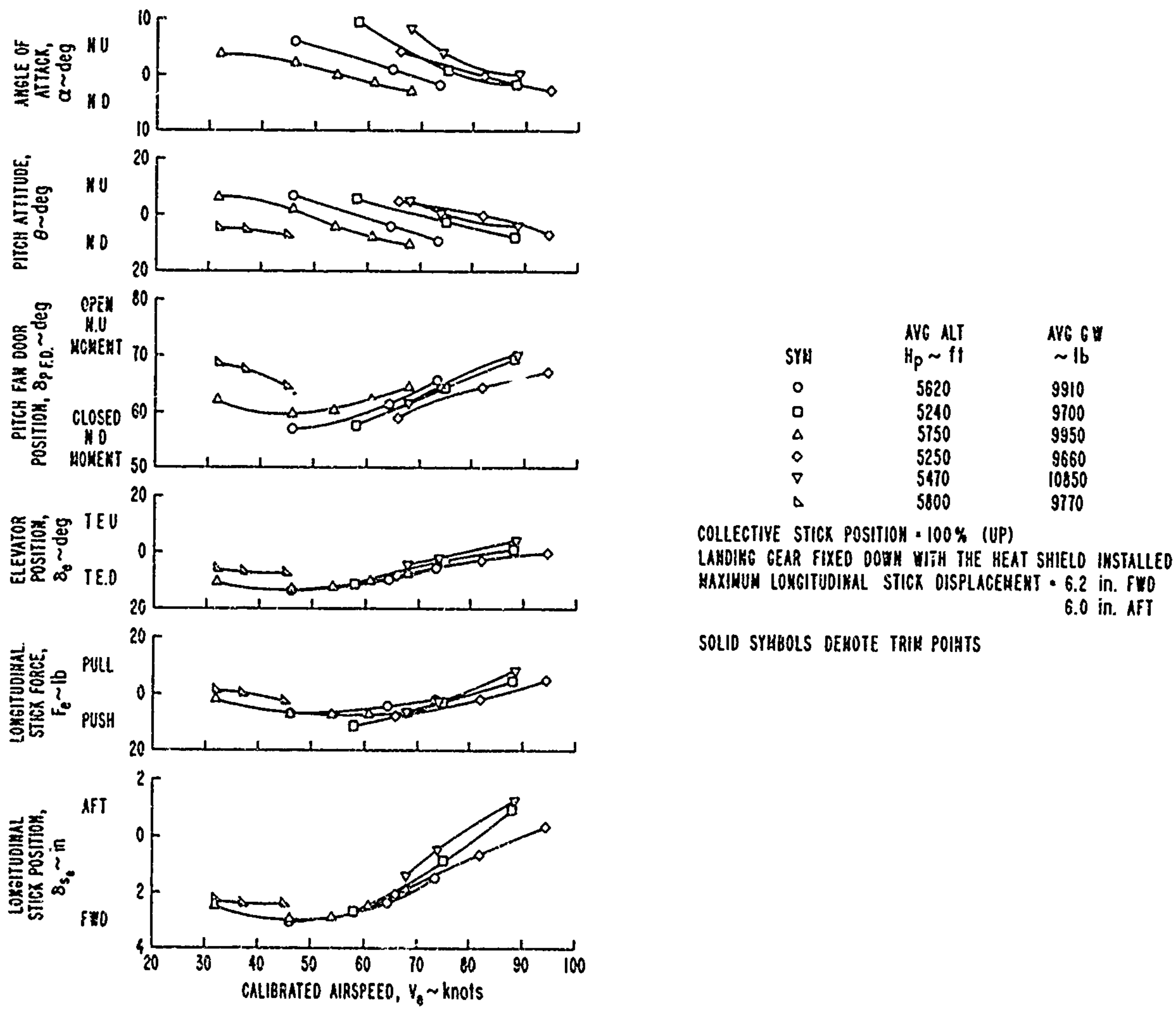


Figure 29. XV-5A Static Longitudinal Stability - Fan Mode.

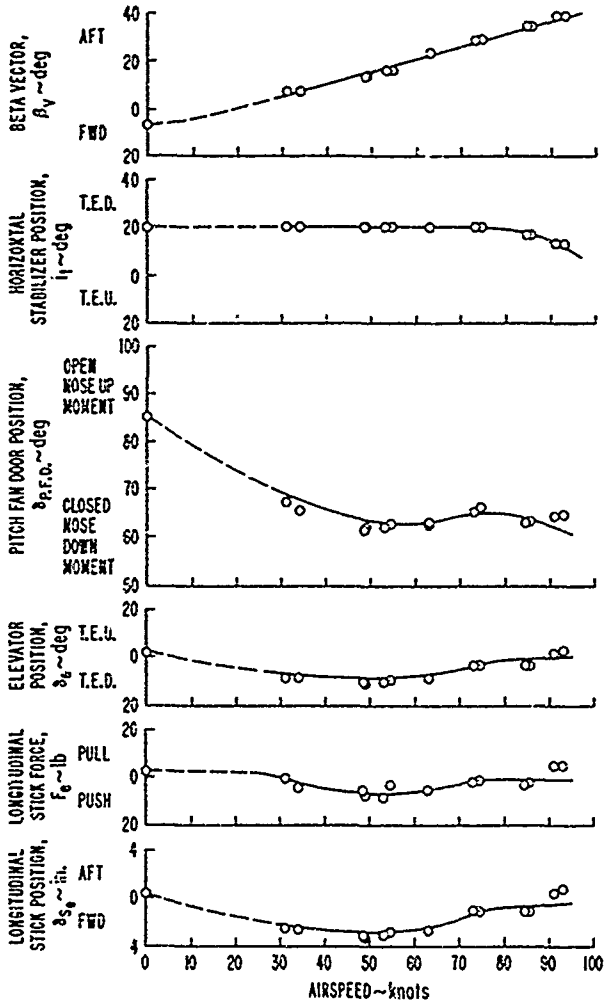


Figure 30. XV-5A Static Longitudinal Trim Stability - Fan Mode.

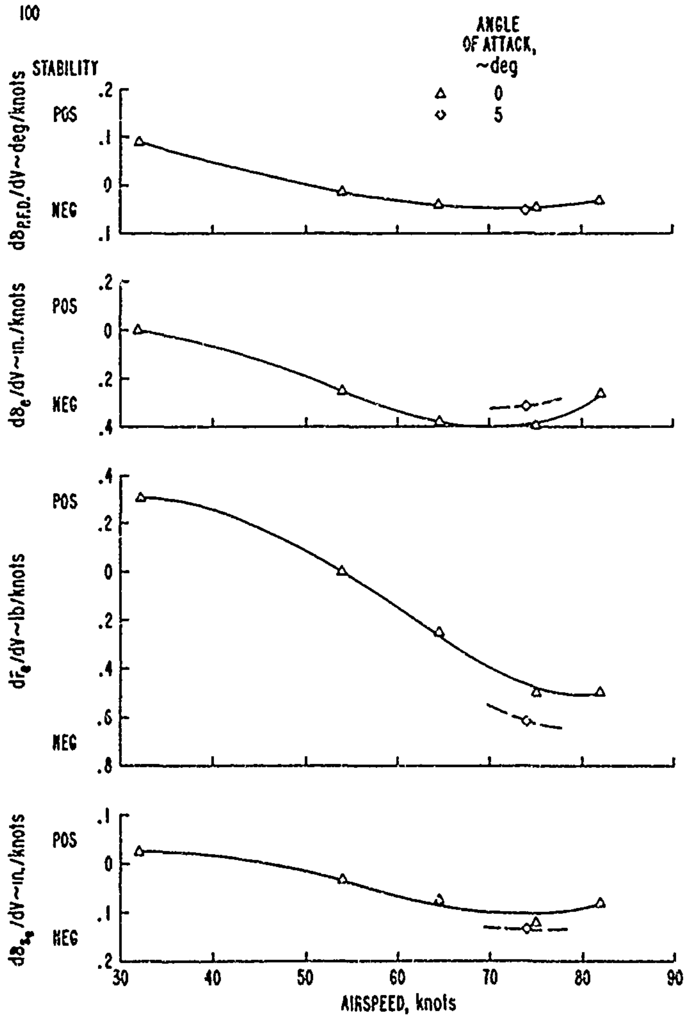


Figure 31. Summary of XV-5A Static Longitudinal Stability - Fan Mode.

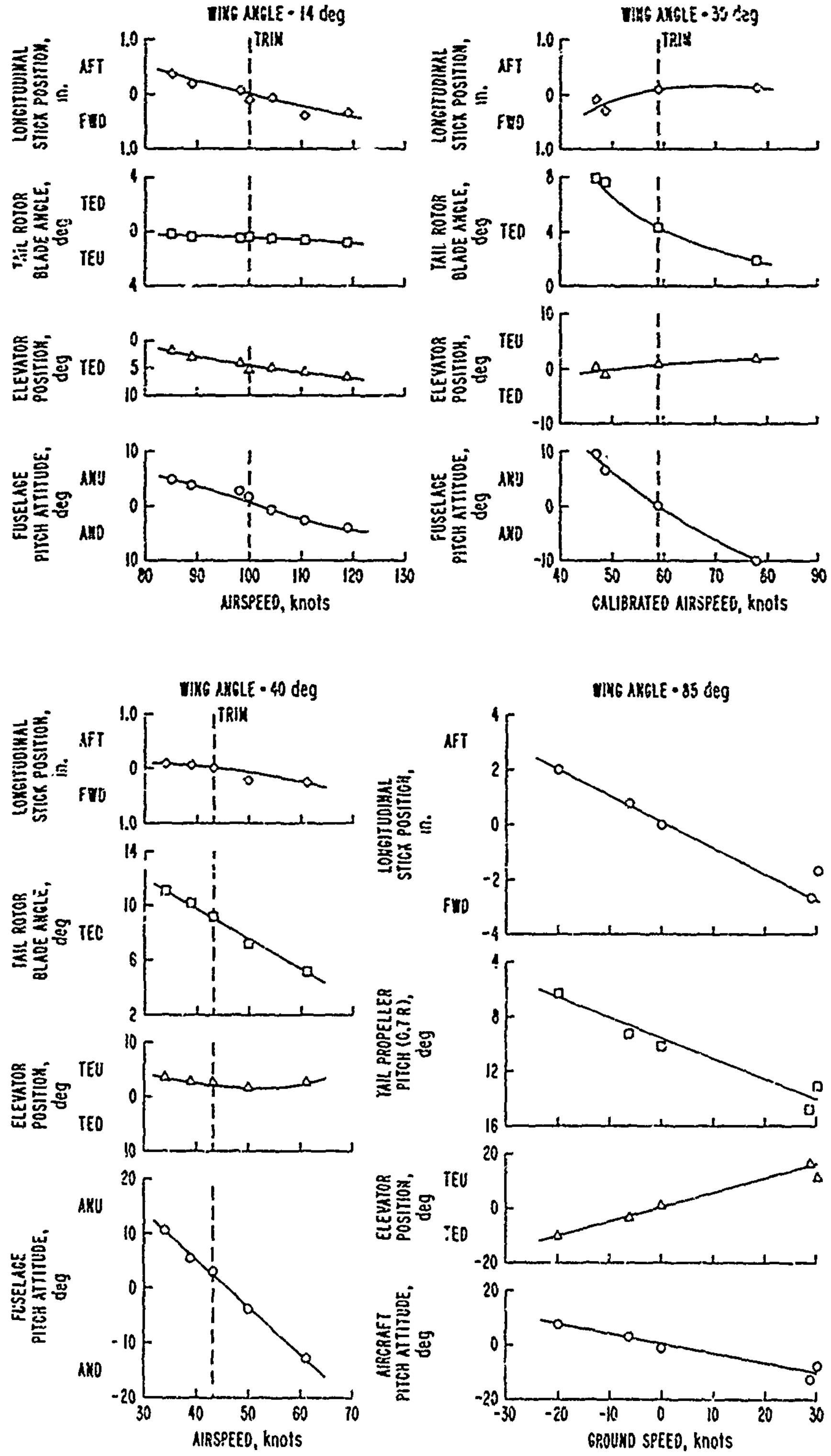


Figure 32. Summary of CL-84 Static Longitudinal Stability - SAS On.

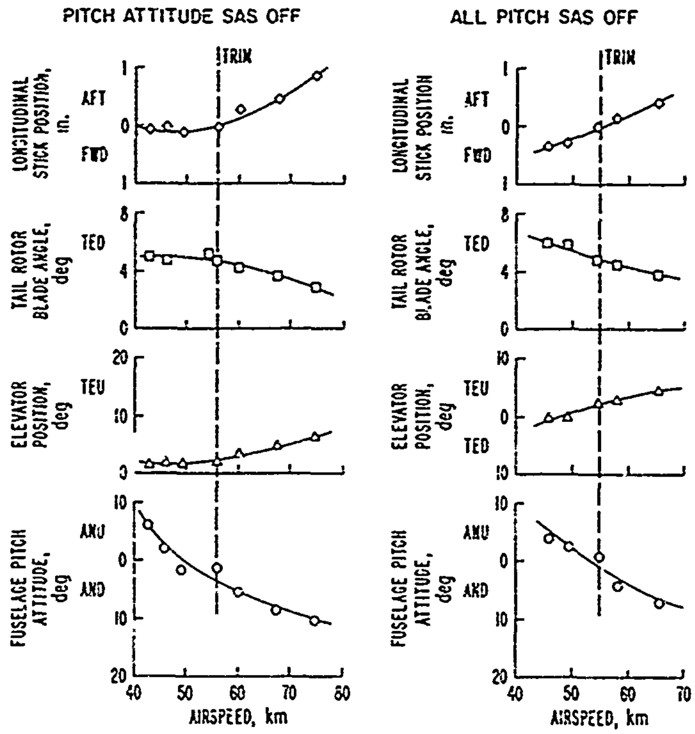


Figure 33. CL-84 Static Longitudinal Stability - Wing Angle = 30°.

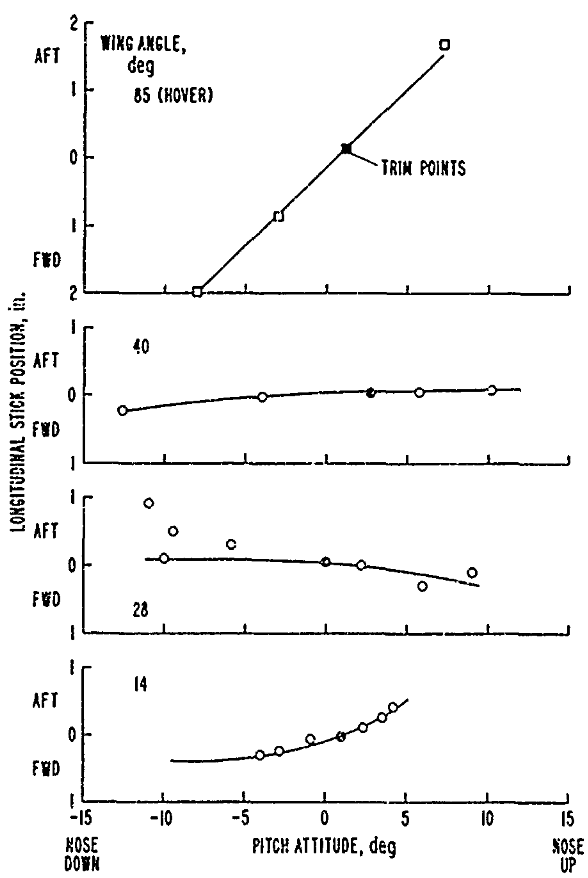


Figure 34. CL-84 Attitude Stability - SAS On (Ref. 6).

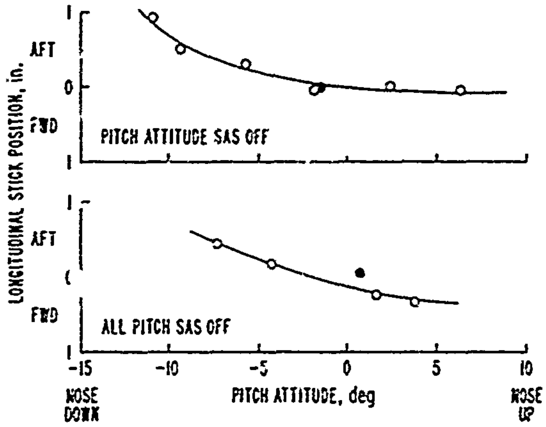


Figure 35. CL-84 Attitude Stability - Wing Angle = 30° (Ref. 6).

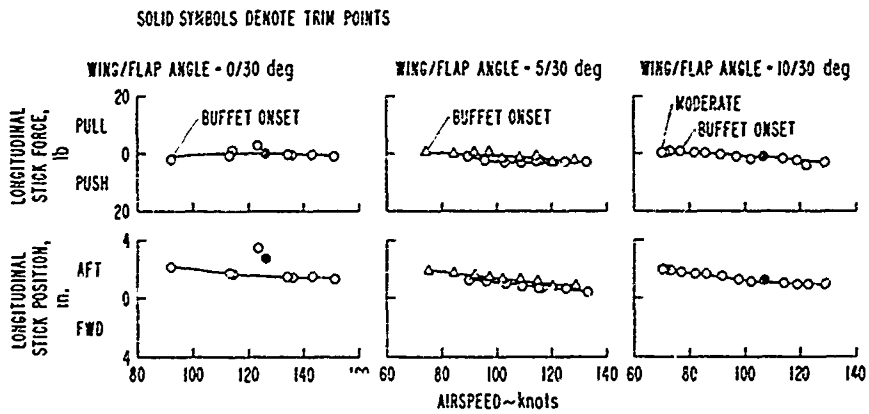


Figure 36. XC-142 Static Longitudinal Stability (Ref. 5).

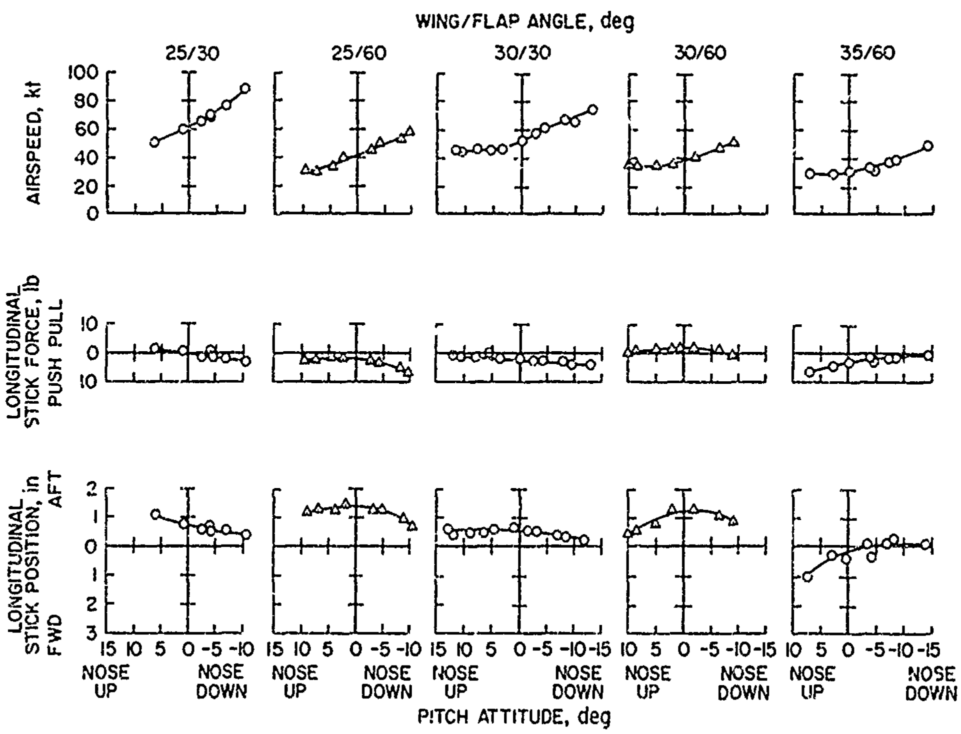


Figure 37. XC-142 Attitude Stability - Pitch Attitude, SAS Off; Pitch Rate, SAS On.

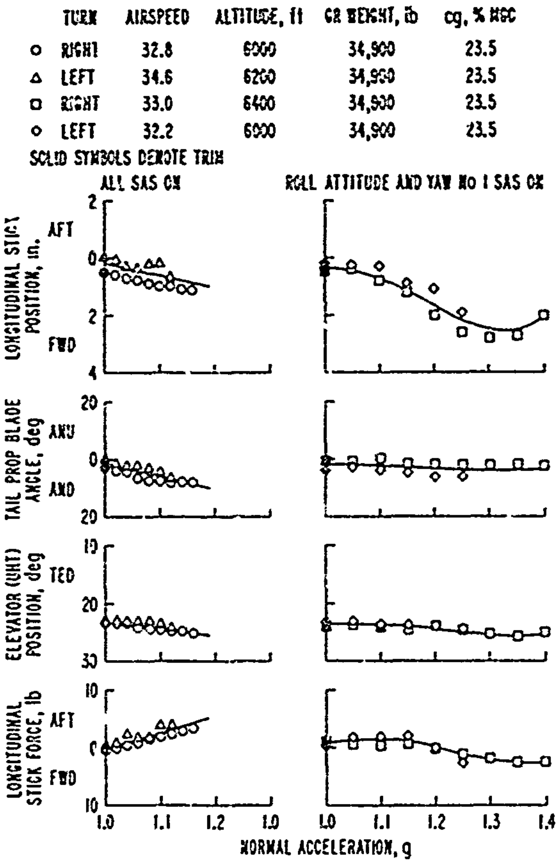


Figure 38. XC-142 Maneuvering Flight Characteristics - Wing/Flap Angle = 35°/60°.

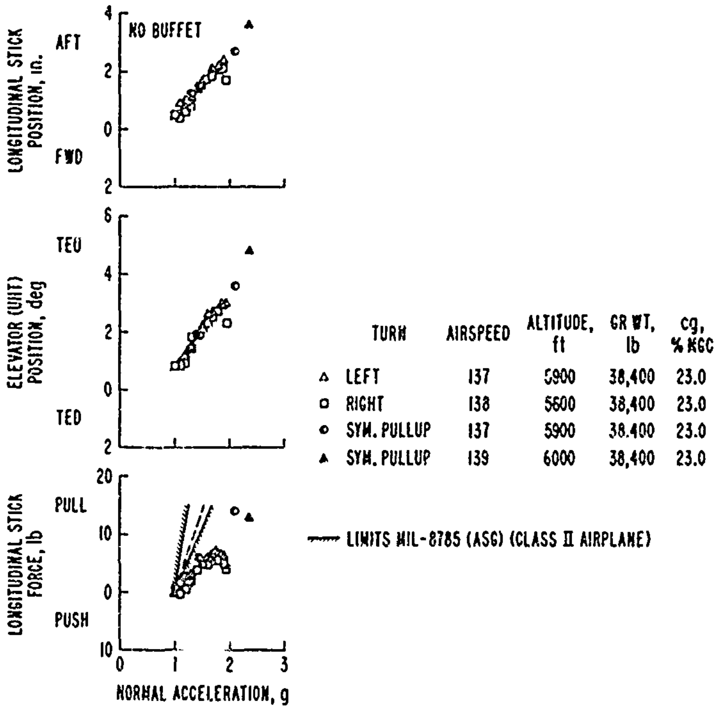


Figure 39. XC-142 Maneuvering Flight Characteristics - Wing/Flap Angle = 0°/30°.

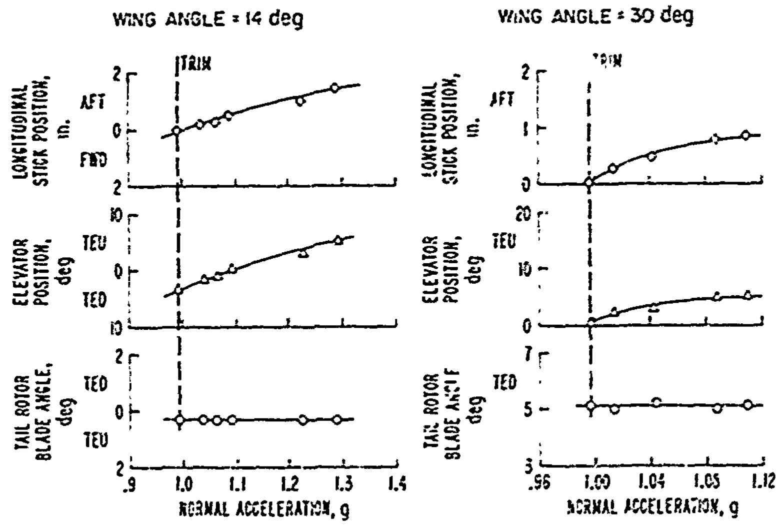
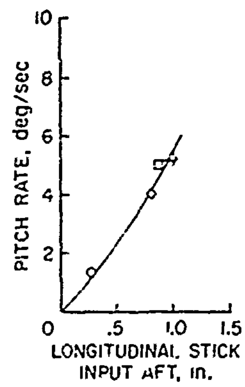


Figure 40. CL-84 Maneuvering Flight Characteristics.



	WING ANGLE, deg	TAIL INCIDENCE, deg	GW, lb	CG, pct
△	14.5	10.1	11,200	29.2
○	30.9	25.7	11,000	27.9
□	41.5	35.5	11,100	29.3
□‾	39.8	34.8	10,900	28.8
◇	84.5	1.0	10,900	38.5

Figure 41. CL-84 Maneuvering Flight Characteristics - (Ref. 6).

	MANEUVER	DUCT ANGLE, deg	AIRSPEED, knots
○	STEADY TURNS	0	118
◇	STEADY PULL-UPS	0	124
△	STEADY PULL-UPS	0	136

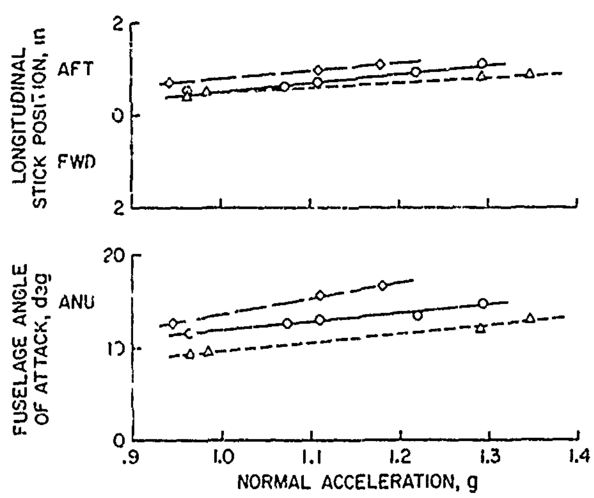


Figure 42. X-22 Maneuvering Flight Characteristics - Duct Angle = 0° (Ref. 19).

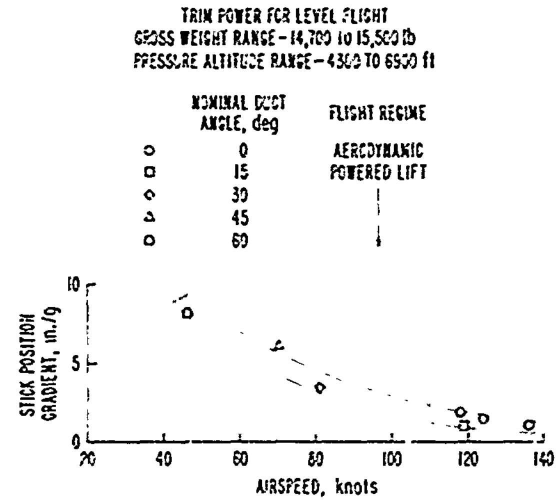


Figure 43. Summary of X-22 Maneuvering Longitudinal Stability.

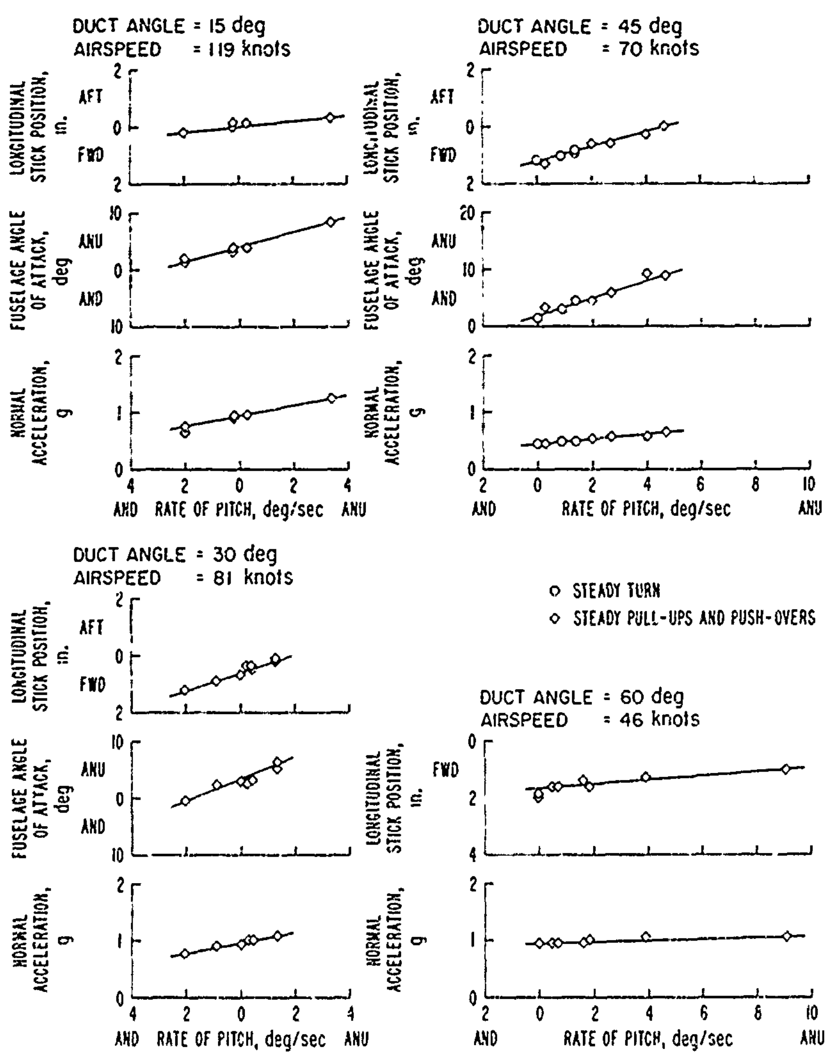


Figure 44. X-22 Longitudinal Maneuvering Flight Characteristics.

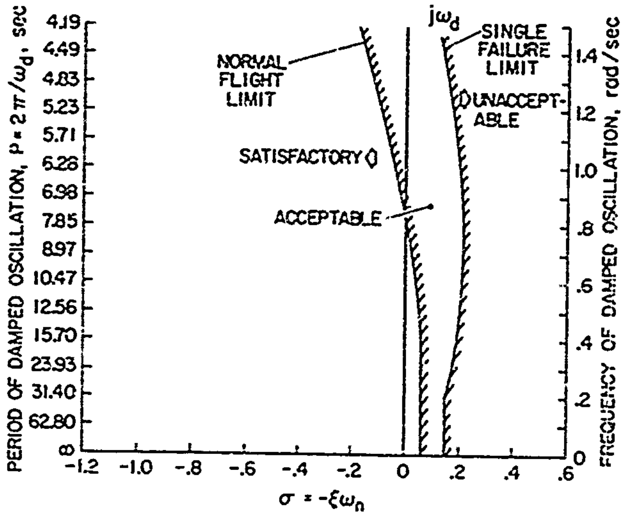


Figure 45. Longitudinal Dynamic Stability Criteria.

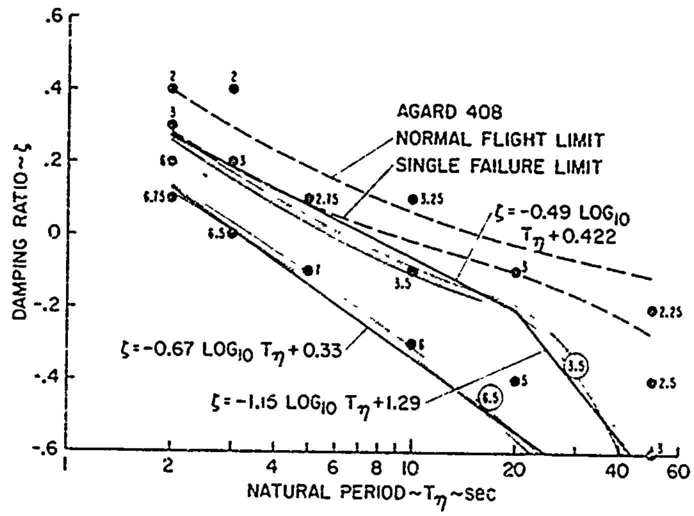


Figure 46. Longitudinal Damping - Period Characteristics for Hover Task.

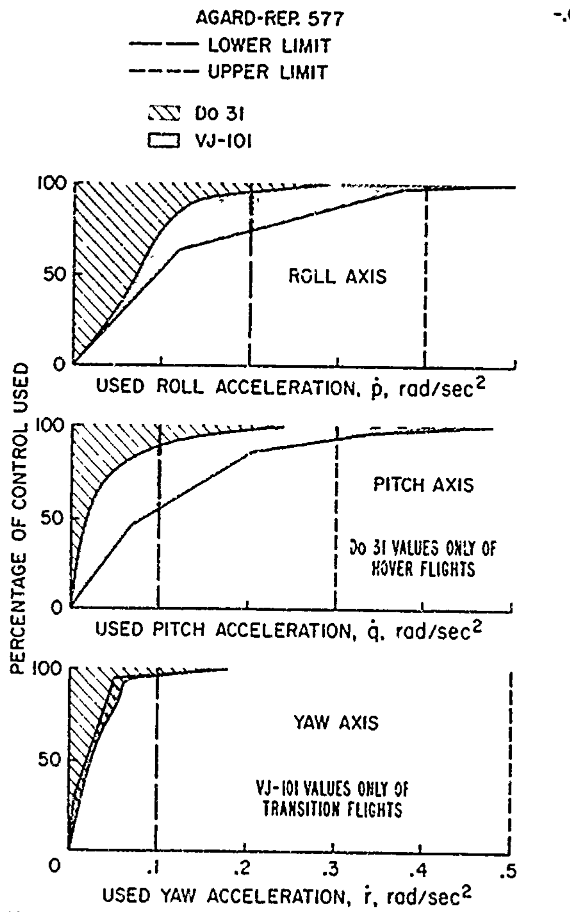


Figure 47. Envelope of Control Acceleration Used in Hover and Transition (Ref. 22).

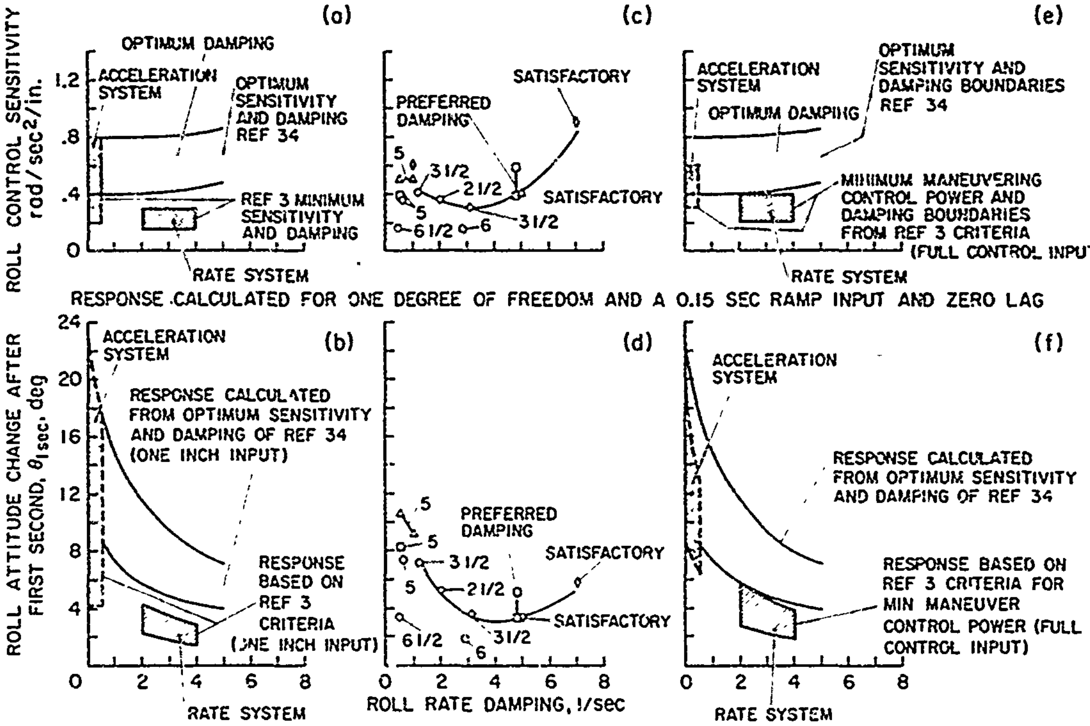


Figure 48. Roll Response Characteristics.

Aircraft	Gross weight (lbs)	I_x moment of inertia (lb-sec^2-ft)	Roll control power (rad/sec^2)	Roll control sensitivity (rad/sec^2/in)	Inherent rate damping (1/sec)	Total rate damping (1/sec)	Flight condition	Comments
VZ-4 tilt-duct	3,200	2,900	0.36	0.06	-0.28	-0.28	Hover	Response was dangerously sluggish
VZ-2 tilt-wing	3,730	1,800	2.92	0.65	-1.0	-1 0	Hover	Maximum roll control required IGE.
X-14A Jet-lift	3,900	2,340	0 8 to 2.05	0.16 to 0 41	-0 45	0 to -2.85	Hover	Basic aircraft had 0.8 control power, -0.45 damping and 6-1/2 pilot rating.
SC-1 jet-lift	6,900	1,865	1.35	0.39 to 0.59	-0.8	-3.7 to 11.9	Hover	Unstabilized aircraft had pilot rating of 4-1/2 - 5. Pilots preferred damping of 3.7 to 4 8 1/sec
SG-1262 hover rig	8,500	1,844	1.5	1.5			Hover	Attitude stabilized ϕ_{max} = 12°, ζ = 0.8, ω_n = 2.68, control power increased from 0.8 to 1.5 from flight test. ϕ_{max} should be > 12°.
XV-5A fan-in-wing	9,200	4,252	1.1	0 31		-3 1	Hover	Response was sluggish when compared to UH-1A Helicopter - pilot rating = 3.5 (damping = 2 0 for
XV-5A fan-in-wing	9,200	4,252	0.92 to 2 1	0.31 to 0 59		-2.0 to -2.7	Transition	Minimum control power at 60 knots. Maximum control power and minimum damping at 125 knots, preconversion mode.
CL-84 tilt-wing	10,600	18,500	1.80	0.36	-0 6	-3.0	Hover	SAS on - sluggish roll response, pilot rating = 4. SAS off - good roll response, little overshoot, P R. = 2.
P1127 jet-lift	12,000	3,900	1 88	0.58	-0.8	-0.8	Hover	Pilot rating of 3.
VJ-101-X1 jet-lift	13,376	32,760	2.25			-2.3	Hover	φ°/in. of stick = 3.6, ω_n = 3.0, ζ = 0.5, ϕ_{max} = ±20°. A second reference shows roll control power = 3 0 rad/sec^2.
BALZAC jet-lift	14,300	4,550	1 12	0.21		-4 5	Hover	
X-22 tilt-duct	14,364	15,029	3 40	0.63 to 0.95	-0.85 to -1.5	-5.0 to -8.0	Hover	High control power required for research role. Sensitivity varied with duct angle and damping varied with sensitivity.
MIRAGE III V jet-lift	26,240	22,200	0.95	0.41		-5 0	Hover	Full control required to trim rolling moments due to side slip.
XC-142 tilt-wing	36,900	191,700	1 2	0.14			Hover	
DO-31 jet-lift	42,300	293,880	0 80 1 24*			-5.0	Hover	ω_n = 2.5, ζ = 1.0, ϕ_{max} = ±18°, φ°/in. of stick = 7 (non linear). 75% roll control required to trim cruise engine failure *1.24 rad/sec^2 from emergency thrust.

Figure 49. Roll Control Response Characteristics of VTOL Aircraft.

Aircraft	Gross weight (lbs)	I_x moment of inertia (lb-sec²-ft)	Roll control power (rad/sec²)	Roll control sensitivity (rad/sec²/in)	Roll rate damping (1/sec)	Time to 30° Bank Angle (sec)	Bank angle in 1 sec (deg)	Airspeed (knots)	Comments
VZ-3RY	2,900	1,440	1.25	0.18	>1.0		17[a]	60	Pilot rating = 4.
CV-48	7,730	6,500	0.7	0.18	0.3	1.6	10[b], 10.5[a]	55-60	Control power pilot rating = 3-1/2. Damping pilot rating = 3-1/2 - 4.
UF-XS	32,000	184,000	0.5	0.04	0.77	2.1	7.0[a]	55	Pilot rating = 3.
C-8A	34,000	273,000[c]	0.6[d]	0.05[d]	1.25	2.1 2.2	6[b] 6.8[a]	70	With spoilers - pilot rating = 5.
C-8A	34,000	273,000[c]	0.3	0.03	1.25	3.4	3.4[a]	70	Without spoilers - pilot rating = 7.
BR941	38,500	225,000	0.42	0.14[c]	1.0	2.4	6.0[b] 5.4[a]	60	1963 tests - pilot rating = 3.
BR941	38,500	225,000	0.48	0.17[c]	0.67	2.2 2.1	4.0[b] 6.8[a]	60	1966 tests - pilot rating = 4 due to control lag.
YC-134	55,500	320,000	0.26[d]	0.03[d]	1.43	1.8	9.6[a]	60	Low sensitivity, undesirable nonlinearity. Pilot rating = 3-1/2.
YC-134	55,500	320,000	0.50	0.03	1.43	2.4	5.6[a]	80	Low sensitivity - pilot rating = 5.
NC-130B	100,000	2,000,000	0.32	0.03	1.11	3.0	4.1[a]	70	Low sensitivity - pilot rating = 5.
NC-130B	100,000	2,570,000	0.43	0.04	1.25	2.6	5.0[a]	85	Low sensitivity - pilot rating = 4.
367-80	150,000	2,570,000	0.55	0.06	1.0	2.0	7.0[a]	85-90	Low sensitivity - pilot rating = 3.
367-80	150,000	2,570,000	0.32		2.78	4.2	4.0[b] 2.9[a]	115	Pilot rating = 2.

[a] Computed assuming 0.1 sec transport lag plus 0.3 sec ramp input and single degree of freedom

[b] Measured from initiation of pilot force

[c] Inertia based on gross weight = 38,000 lb

[d] Nonlinear with deflection, sensitivity measured at 0 to 1 inch control deflection

Figure 50. Roll Control Response Characteristics of STOL Aircraft.

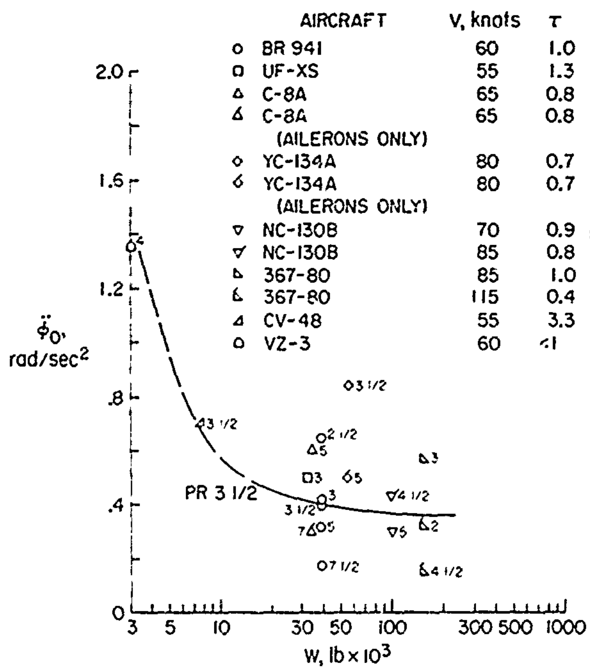


Figure 51. Roll Control Power of Various STOL Aircraft.

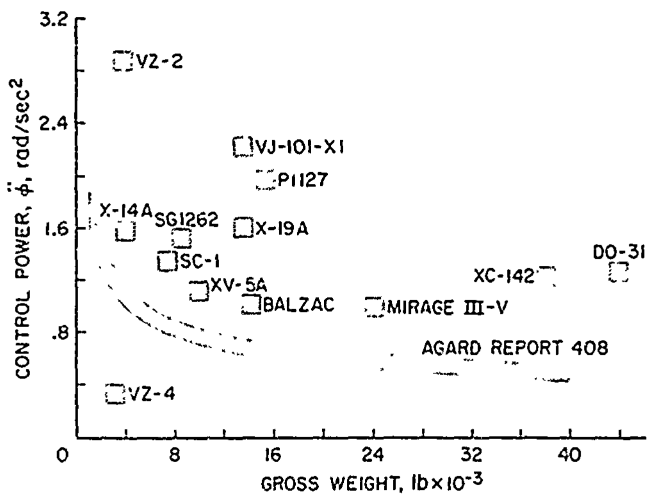


Figure 52. Roll Control Power of Various VTOL Aircraft.

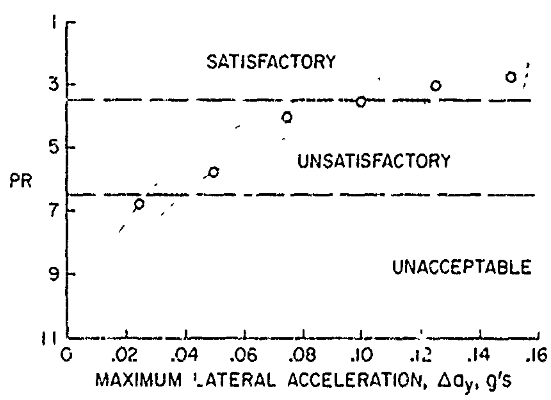


Figure 53. Effect of Lateral Acceleration on Pilot Rating, X-14A Aircraft (Ref. 30).

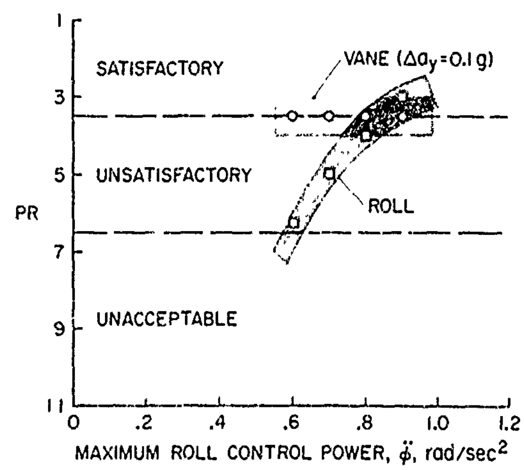


Figure 54. Comparison of Vane Translation and Conventional Roll Control Methods for Lateral Maneuvering, X-14A (Ref. 30).

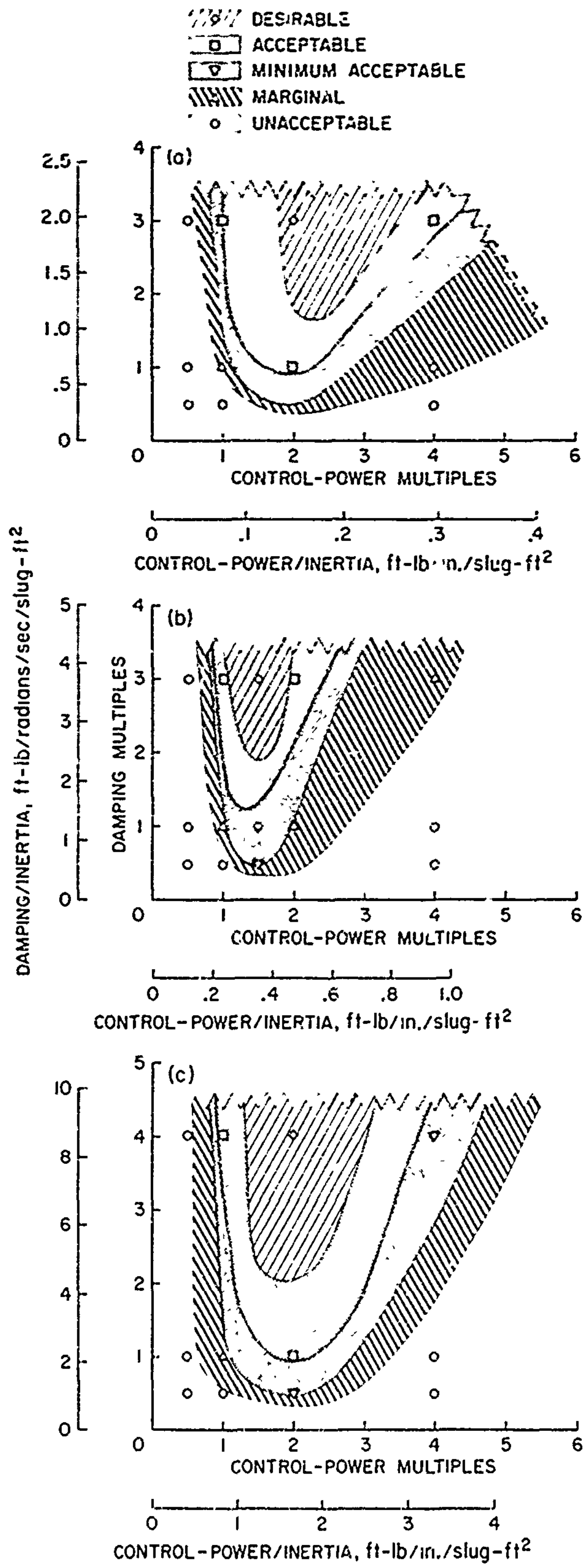


Figure 55. Handling-Qualities Boundaries as a Function of Damping and Control Power (Ref. 30) - (a) Pitch Axis; (b) Roll Axis; (c) Yaw Axis.

Damping, 1/sec	Control power, radians/sec²									Pilot
	2			1.4			0.8			
	Sensitivity, radians/sec²/in.									
	0.45	0.59	0.73	0.31	0.40	0.51	0.18	0.23	0.29	
-2.9	2-3	2-1/2	2-1/2	4-5	4-5	4	6	7	6	A
	3	2-1/2	3-1/2	4	3-1/2	4-1/2	5-1/2	6	5-1/2	B
-1.75	3	3	3	4-5	5	4-1/2	5-1/2	6	5-1/2	A
	4	4	3-1/2	4-1/2	4-1/2	5	4-1/2 - 5	6-1/2	6	B
-.55	3	2-1/2	3	4	4	5	5	5	7	A
	4-1/2	4-1/2	4	5-1/2	5-1/2	5-1/2	6-1/2	7	7	B

Figure 56. Effect of Roll Control Sensitivity and Damping on Pilot Rating (Ref. 32).

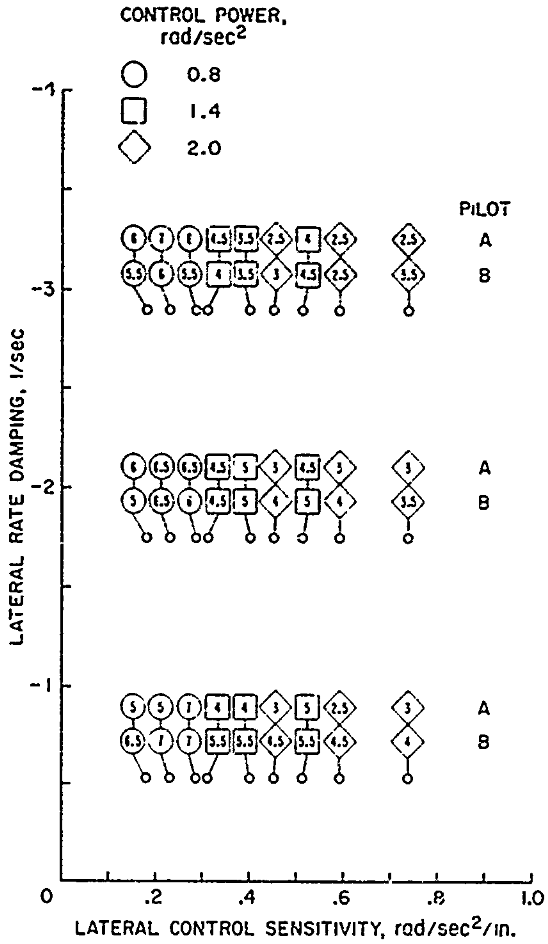


Figure 57. Effect of Roll Control Sensitivity and Damping of Pilot Rating for Various Values of Roll Control Power (Ref. 32).

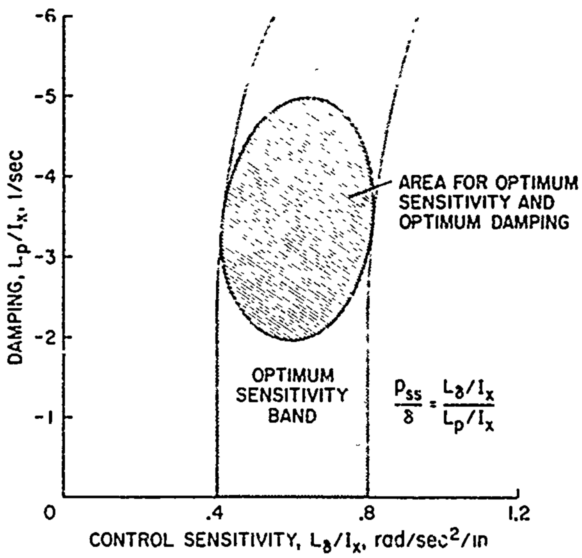


Figure 58. Effect of Control Sensitivity on Pilot Rating - Acceleration Control System (Ref. 34).

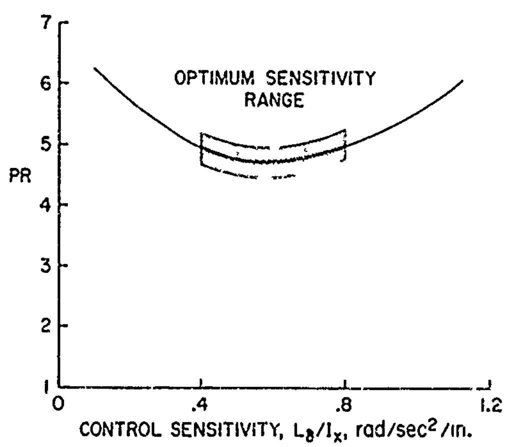


Figure 59. Variation of Optimum Control Sensitivity With Damping (Including Range of Optimum Damping) (Ref. 34).

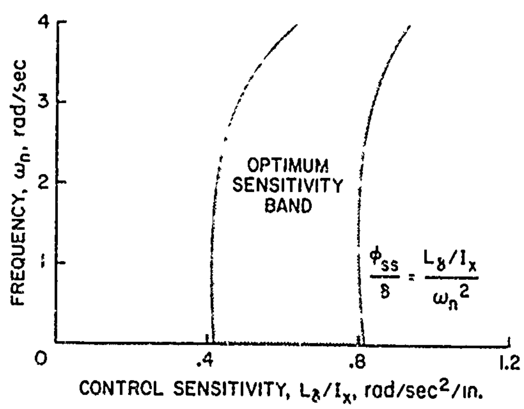


Figure 60. Variation of Optimum Control Sensitivity With Frequency Attitude System (Ref. 34).

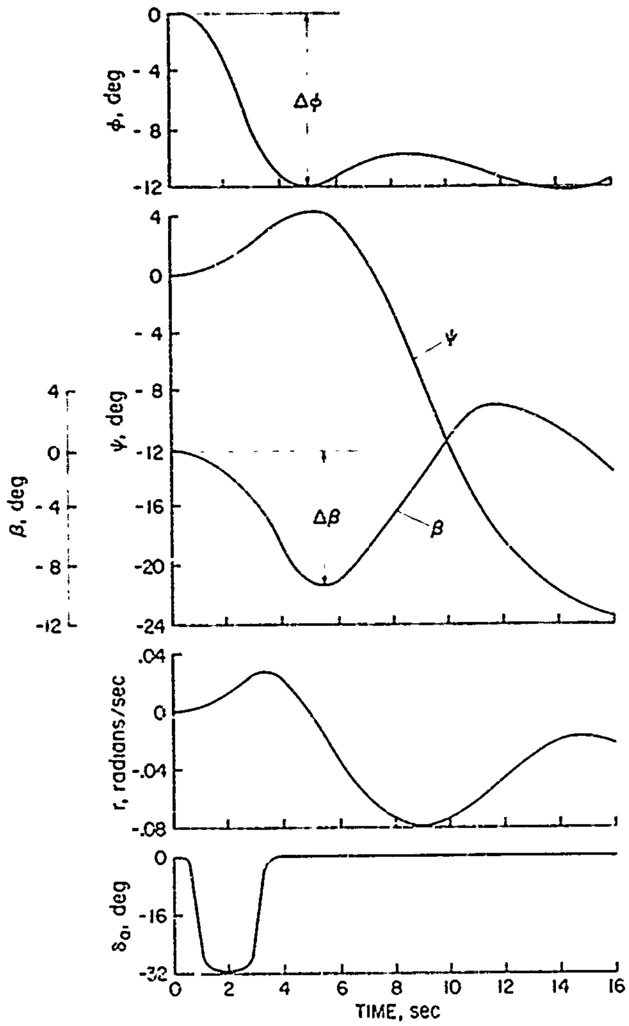


Figure 61. Time History of the Response of the NC-130B Aircraft to a Step Bank Maneuver; V = 70 knots (Ref. 4).

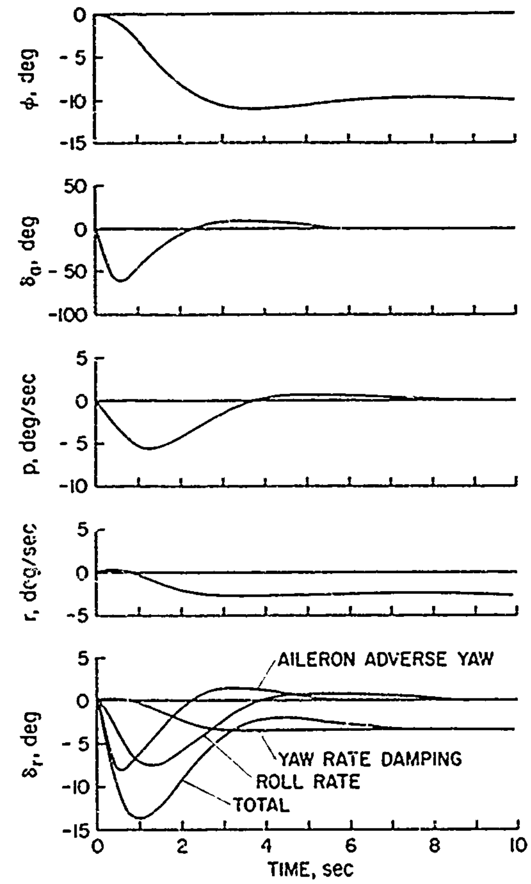


Figure 62. Time History Showing Rudder Requirements for a Coordinated (β = 0) Turn Maneuver; V = 70 knots, NC-130B Aircraft (Ref. 36).

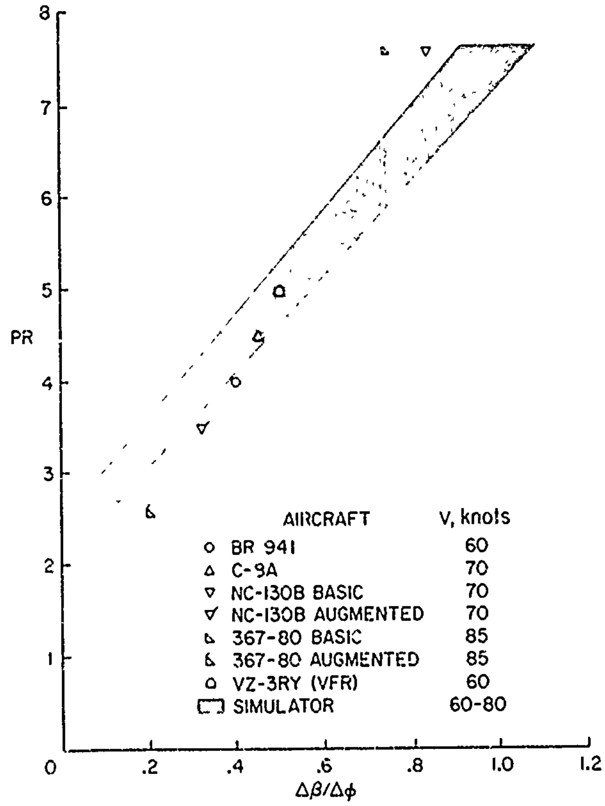


Figure 63. Relation of Turn Entry Coordination and Pilot Opinion in IFR Conditions (Ref. 4).

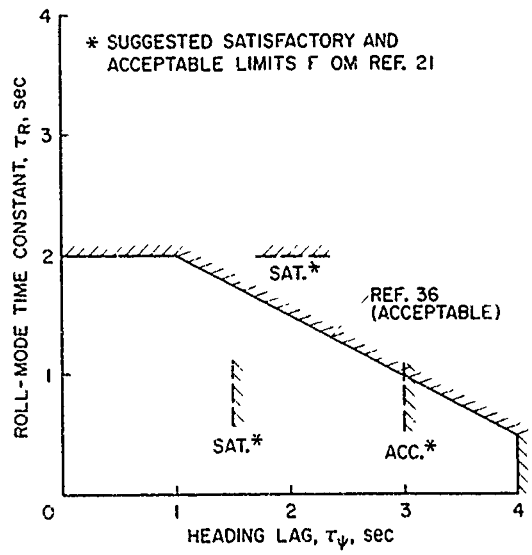


Figure 64. Roll Mode Time Constant and Heading Lag Requirements (Ref. 38).

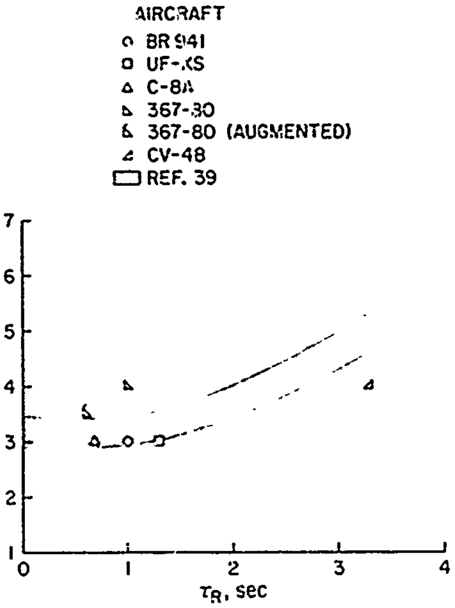


Figure 65. Variation of Pilot Rating With Roll Time Constant (Ref. 4).

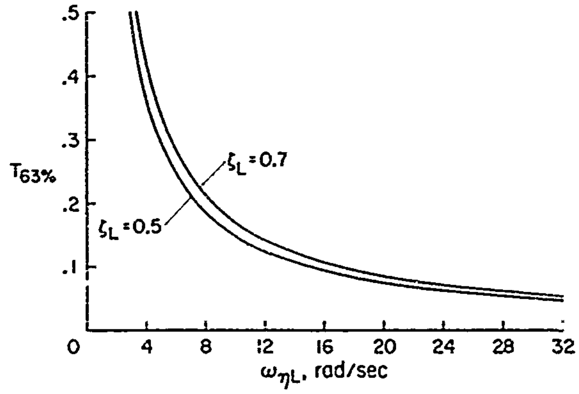


Figure 66. Variation of $T_{63\%}$ and ω_{n_L} for Various Second-Order System Parameters.

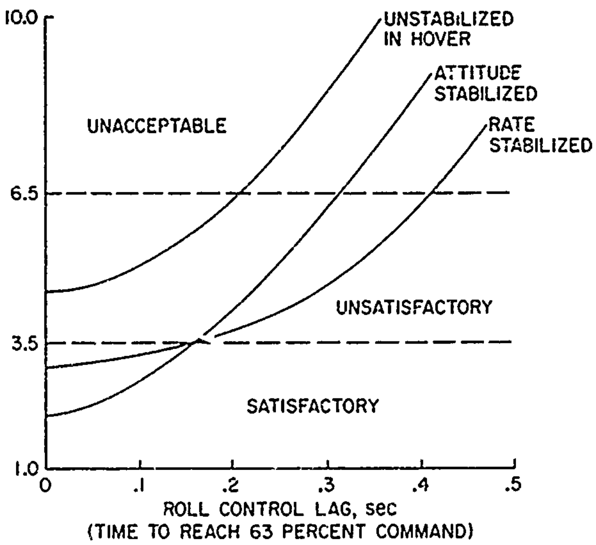


Figure 67. Effect of First-Order Lag in Roll Control.

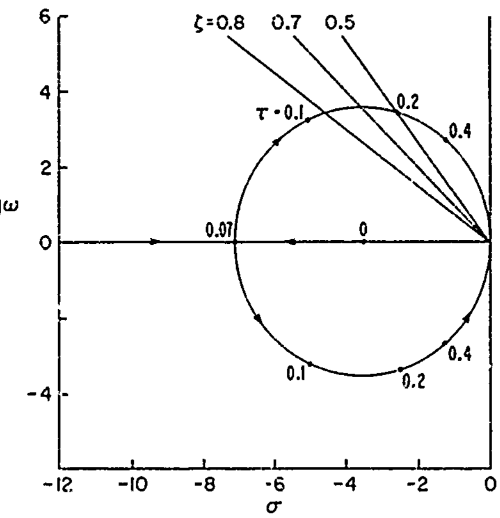


Figure 68. Pole-Zero Map for a Rate-Stabilizer Control System With First-Order Control Moment Lag.

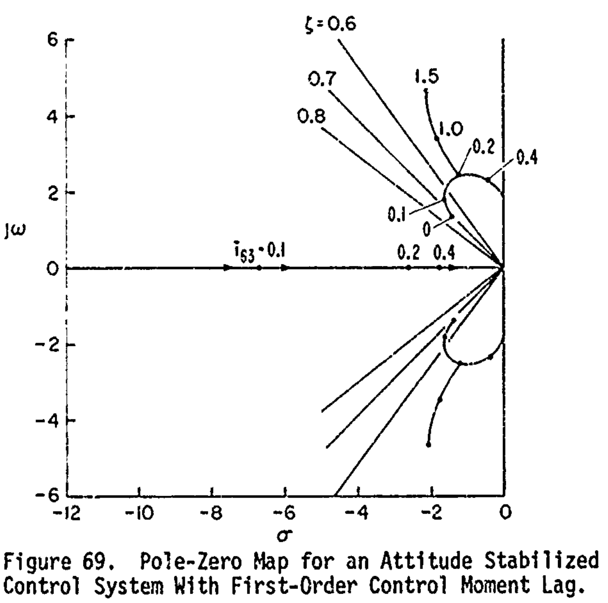


Figure 69. Pole-Zero Map for an Attitude Stabilized Control System With First-Order Control Moment Lag.

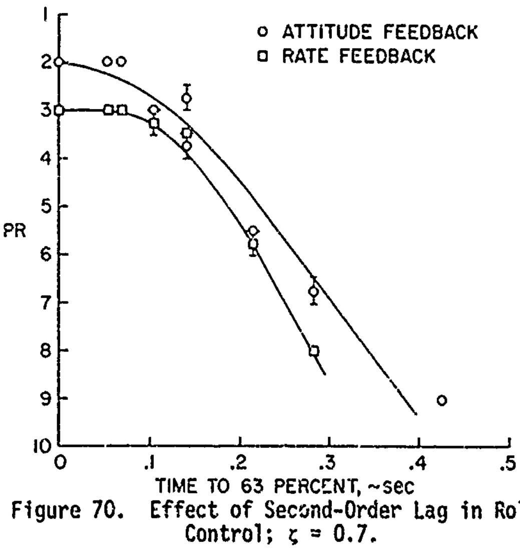


Figure 70. Effect of Second-Order Lag in Roll Control; $\zeta = 0.7$.

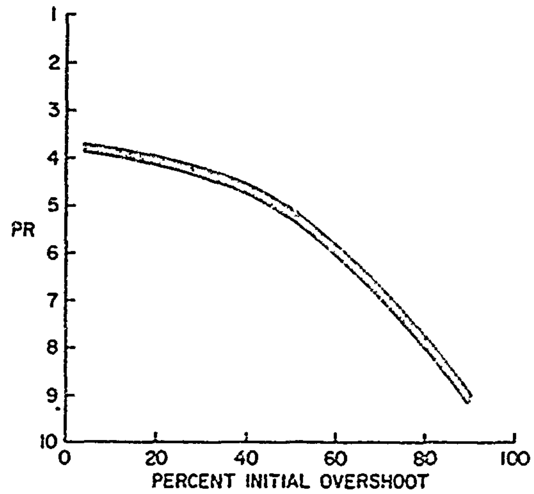


Figure 71. Effect of Overshoot With Second-Order Control Lag.

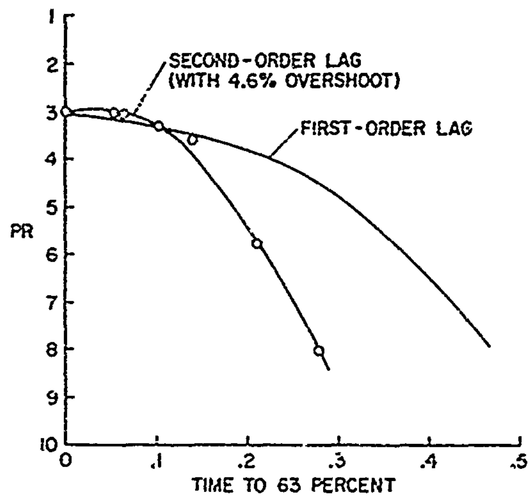


Figure 72. Comparison of First- and Second-Order Lag Effect on Roll Control for a Rate-Damped System.

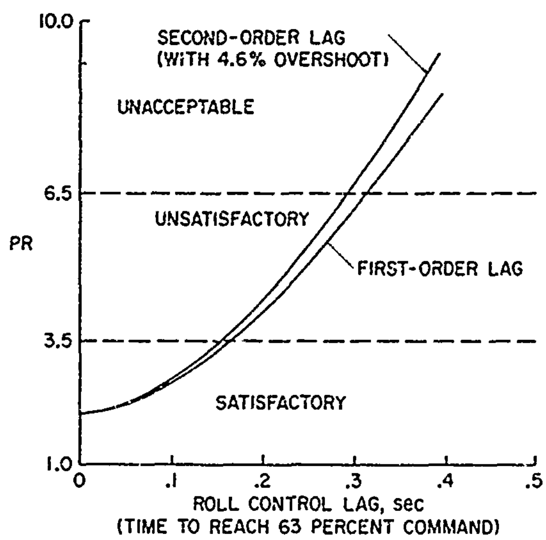


Figure 73. Comparison of First- and Second-Order Lag Effects on Roll Control for an Attitude-Stabilized System.

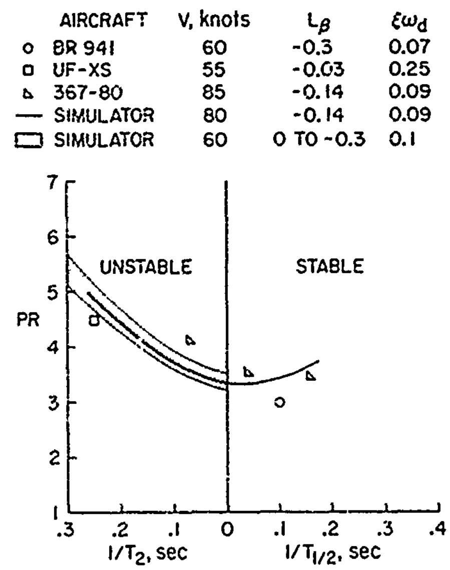


Figure 74. Variation of Pilot Rating With Spiral Stability (Ref. 4).

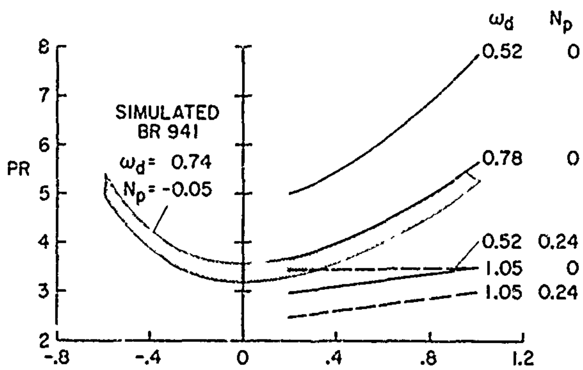


Figure 75. Effect of Dihedral on Pilot Rating (Ref. 4).

Aircraft	Gross weight (lbs)	I_{zz} moment of inertia (lb-sec²-ft)	Yaw control power (rad/sec²)	Yaw control sensitivity (rad/sec²/in)	Inherent rate damping (1/sec)	Total rate damping (1/sec)	Flight condition	Comments
VZ-4 tilt-duct	3,200	3,100	0.70	0.2			Hover	Correcting for a yawing velocity required full control and was not enough to prevent turning 90°.
VZ-2 tilt-wing	3,700	4,370	36 to 0.95	0.12 to 0.32	-0.5	-0.5	Hover	Control power generated by ailerons and/or tail fan. Pilots objected to nonlinear response. Control power was insufficient in ground effect
X-14A Jet-lift	3,950	5,400	0.35 0.60	0.12 0.20	-0.2 -0.2	-0.2 0.95	Hover	Pilot ratings 7.5, 4.5 and 5.5 } 3 pilots Pilot ratings 3.0, 3.0 and 3.0 }
SC-1 Jet-lift	6,900	7,000	0.35	0.12	-0.1	-0.1	Hover	Control power was low for precise heading hold.
SG-1262 Power-rig	8,585	25,521	0.16	0.12			Hover	Low control power, pilot rating = 5. Time to ψ_{max} = 3.0 sec, full opposite pedal required 30° before desired heading
XV-5A fan-in-wing	9,200	17,420	0.36 to 0.43	0.10 to 0.12		-1.2 to -1.7	Hover	Response is function of height control position. High collective setting = high response
XV-5A fan-in-wing	9,200	17,420	0.49 to 0.67	0.14 to 0.19		-1.6 to -2.2	Transition	Minimum control power and minimum damping at 30 knots. Maximum control power and maximum damping at 125 knots
CL-84 tilt-wing	10,600	32,600	0.47	0.21	-0.4	-2.2	Hover	Nonlinear response - large control deflection required to get desired response. Pilot rating = 5. Good trim capability 10% control remaining at 60° wing angle with 22-31 knots wind
P1127 Jet-lift	12,000	21,400	0.38	0.25	-0.2	-0.2	Hover	Full control power obtained in 75% of control travel.
VJ-101-X1 Jet-lift	13,375	45,420	0.95	0.21		-0.9	Hover	A second reference shows the yaw control power to be only 0.40 rad/sec²
BALZAC Jet-lift	14,300	30,700	0.11			-0.55	Hover	Control power too low.
X-22 Tilt-duct	14,365	45,260	0.70	0.25		-0.5 to -2.0	Hover	Pilot rating = 3 for sensitivity and damping.
MIRAGE III V Jet-lift	26,240	210,000	0.18	0.		-1.40	Hover	
XC 142 tilt-wing	36,900	270,000					Hover	
DO-31 Jet-lift	42,300	454,324	0.38 0.48*				Hover	Control power from emergency thrust.

Figure 76. Yaw Control Response Characteristics of VTOL Aircraft.

Aircraft	Gross weight (lbs)	I_{zz} moment of inertia (lb-sec²-ft)	Yaw control power (rad/sec²)	Yaw control sensitivity (rad/sec²/in)	Yaw rate damping (1/sec)	Time for 15° heading change (sec)	(a) Heading change in 1 sec (deg)	Maximum steady sideslip angle (deg)	Airspeed (knots)	Comments
VZ-3RY	2,900	3,450	<0.3						60	Pilot rating = 6.
CV-48	7,730	12,300	0.33	0.10	0.7	1.7	4.6 7.0(b)	16(c)	55-60	Pilot rating = 3. Pilot wanted a little more. Maximum sideslip angle (+ > 16°).
UF-XS	32,000	329,000	0.07 left 0.27 right	0.07	0.5	3.9 left 1.8 right	1.0 left 3.9 right	22(d) left 16 right	5	(Left) Pilot rating = 5. (Right) Pilot rating = 3. True sideslip angle believed to be 2/3 of indicated value.
C-8A	34,000	447,000	0.32	0.16	0.34				70	Pilot rating = 3.
BR941	38,500	400,000	0.18	0.06	0.15	2.0	3.0	25	60	
YC-134A	55,500	520,000						12(e)	70	Unsatisfactory control lightening at sideslip angles >10°
NC-130B	100,000	2,800,000	0.21	0.06	0.12	1.9	3.3	18(f)	70	Pilot rating = 4. Much higher sideslip angles were possible but concerned about fin stall.
NC-130B	100,000	2,800,000	0.32	0.09	0.15	1.5	5.1		85	Pilot rating = 3.5.
367-80	150,000	4,730,000	0.09		-0.16	2.7	1.5	18	85-90	Pilot rating = 3. (Sensitivity a little high).
367-80	150,000	4,730,000	0.12		-0.16	2.4	2.0		115	Pilot rating = 2.5.

*Computed for single degree of freedom with 0.1 sec transport lag plus 0.3 sec control ramp.

Figure 77. Yaw Control Response Characteristics of STOL Aircraft.

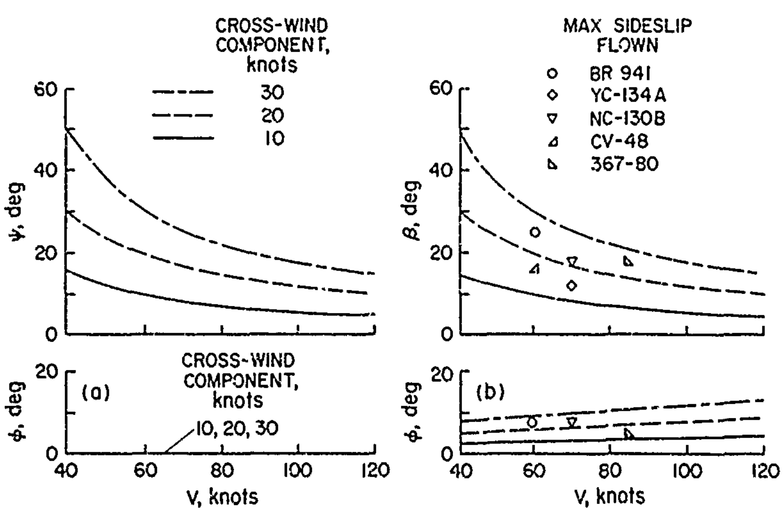


Figure 78. Effect of Airspeed on Cross-Wing Approaches (Ref. 4).

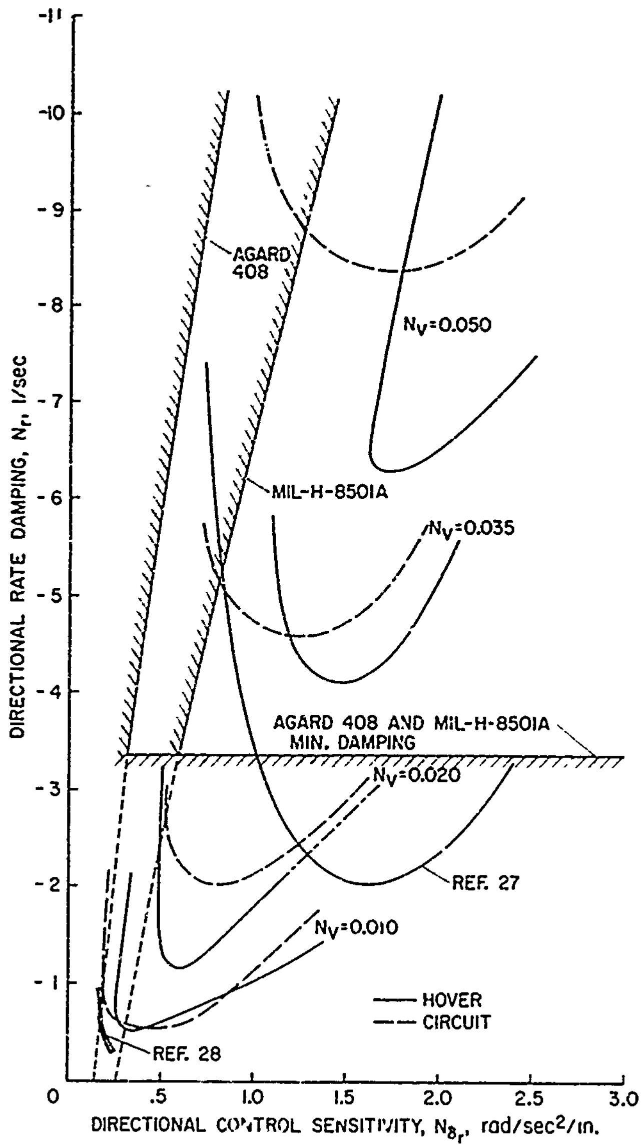


Figure 79. Comparison of Three and One-Half Boundaries for Various Values of Yawing Moment Due to Side Velocity, N_V (Ref. 50).

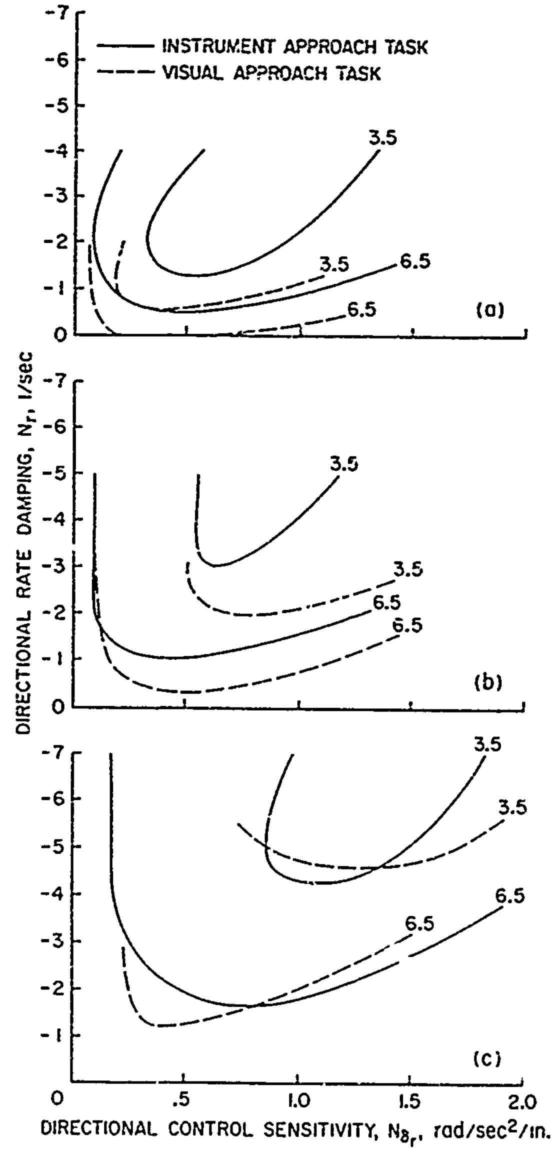


Figure 80. Comparison of Handling Qualities Boundaries for Visual and Instrument Approach Tasks (Ref. 51).

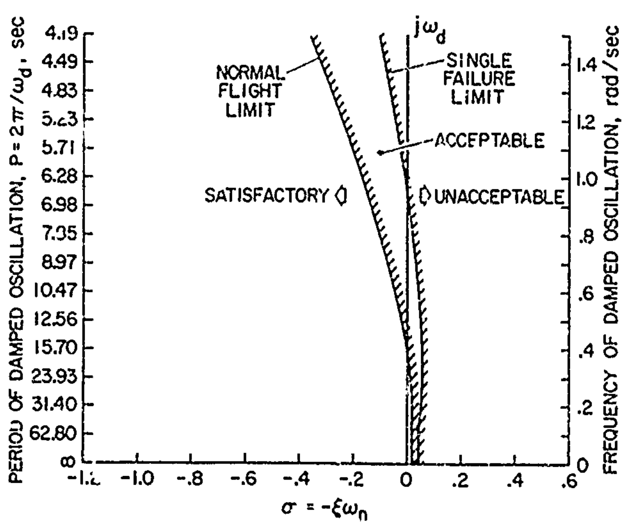


Figure 81. Lateral-Directional Dynamic Stability Criteria.

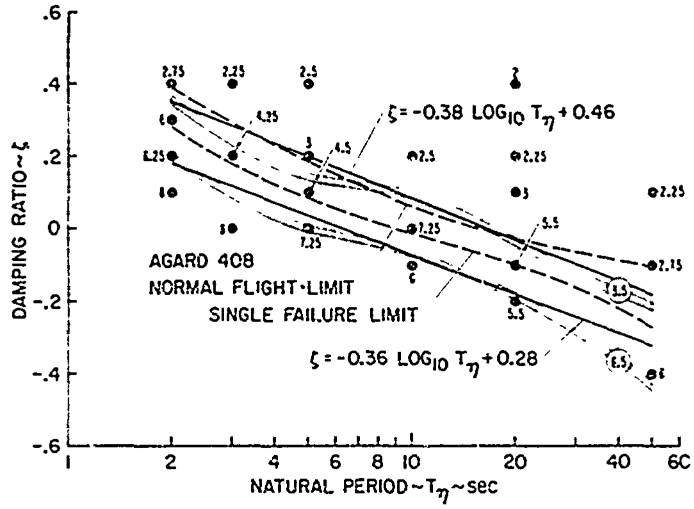


Figure 82. Lateral-Directional Damping Period Characteristics for Hover Tasks.

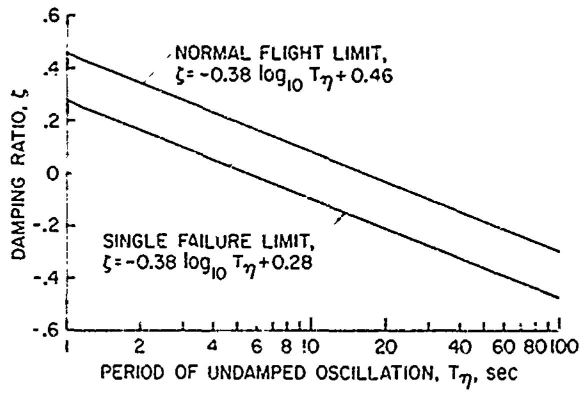


Figure 83. Lateral-Directional Dynamic Stability Boundaries.

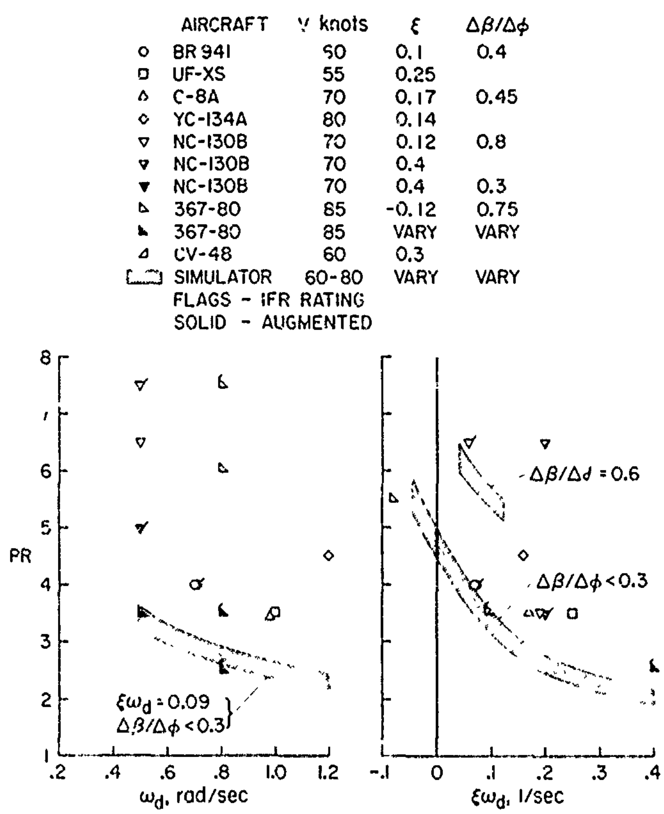


Figure 84. Pilot Opinion of Directional Frequency and Damping for Several STOL Aircraft (Ref. 4).

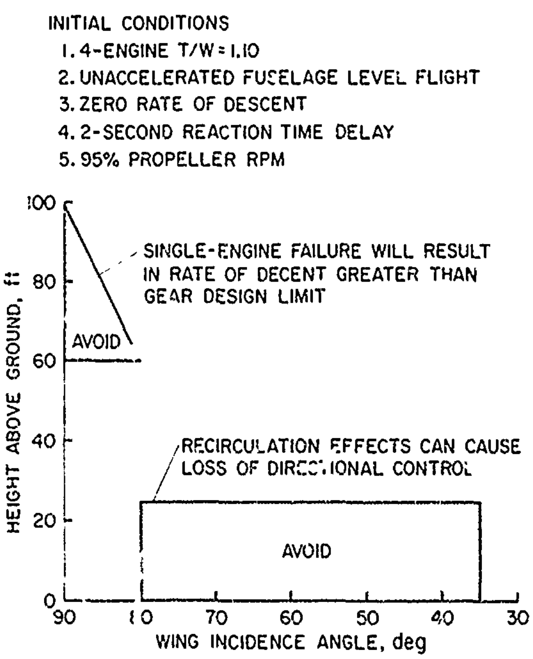


Figure 85. Single Engine Failure Boundaries for XC-142 Aircraft.

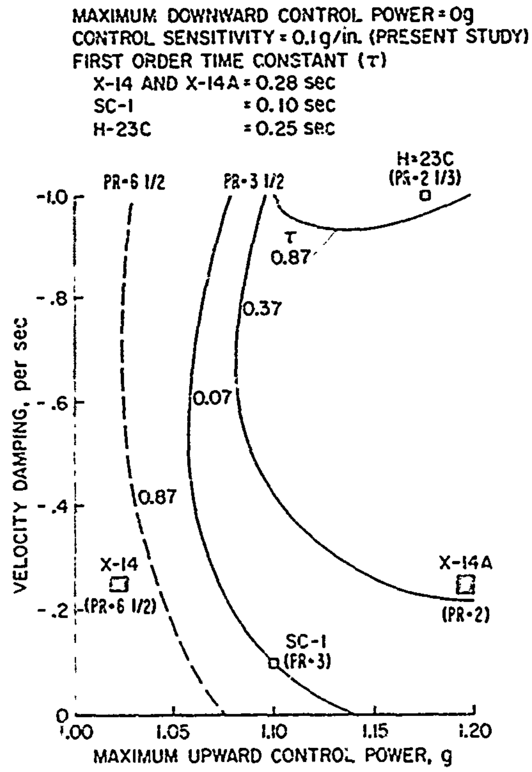


Figure 86. Maximum Control Power Boundaries Out-of-Ground Effect at Various Levels of Time Constant (Ref. 54).

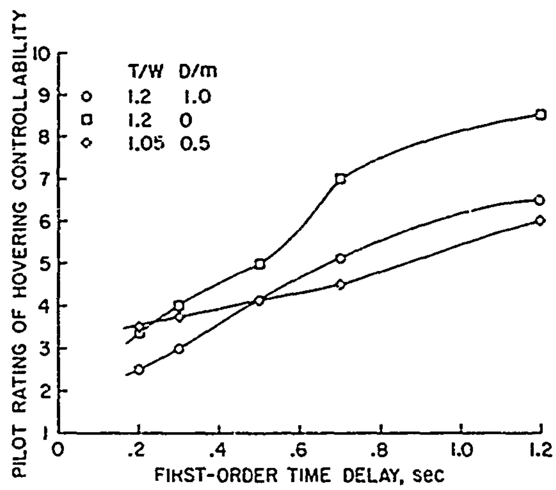


Figure 87. Variation in Pilot Rating of Overall Controllability Due to Control-System Time Delay Out-of-Ground Effect (Ref. 51).

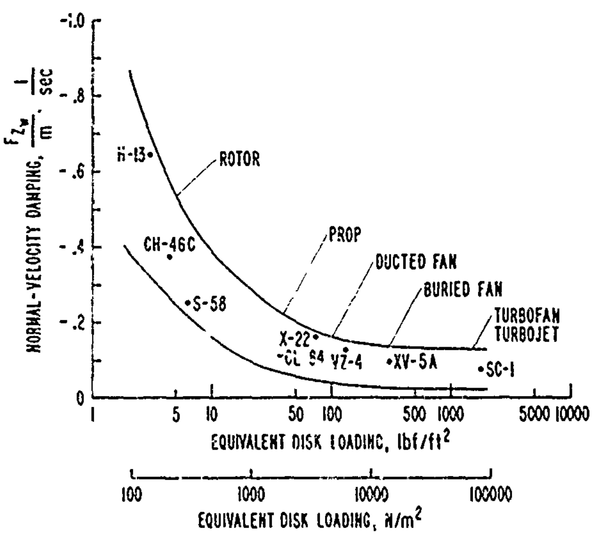


Figure 88. Variation of Normal-Velocity Damping With Disk Loading (Ref. 56).

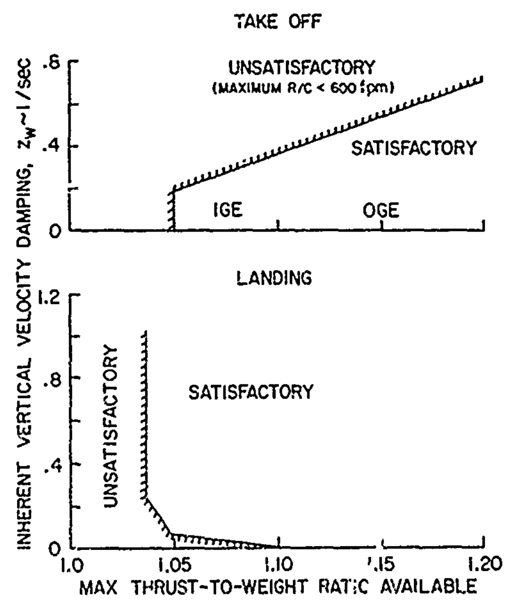


Figure 89. Vertical Height Control Characteristics.

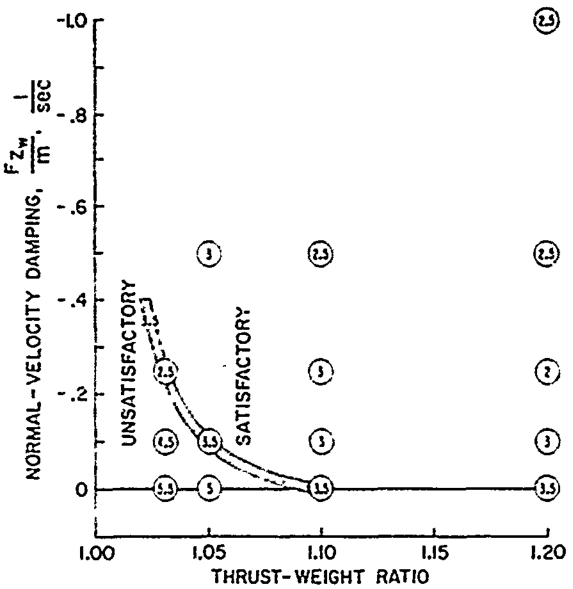


Figure 90. Combinations of Thrust-Weight Ratio and Damping Required to Provide a Capability for Satisfactorily Arresting the Descent Rates (Ref. 56).

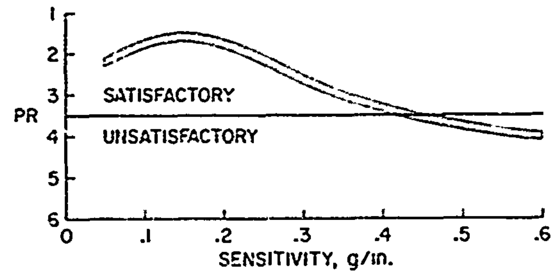


Figure 91. Variation of Pilot Rating With Control Sensitivity (Ref. 56).

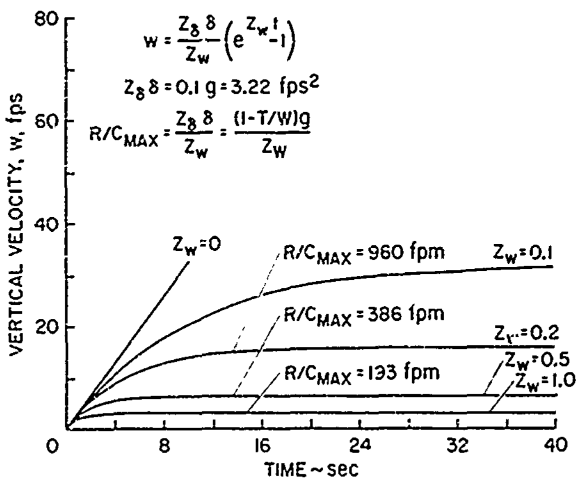


Figure 92. Effect of Velocity Damping on Vertical Velocity With Control Sensitivity of 0.1 g/in. (Zero Control Lag).

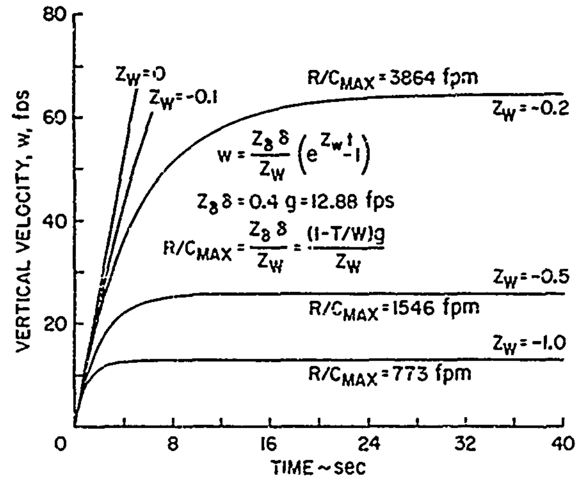


Figure 93. Effect of Velocity Damping on Vertical Velocity With Control Sensitivity of 0.4 g/in. (Zero Control Lag).

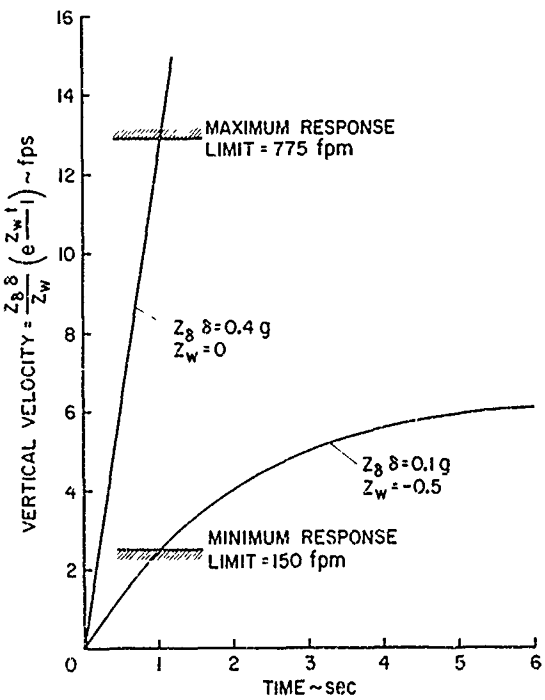


Figure 94. Boundaries of Thrust Response Characteristics.

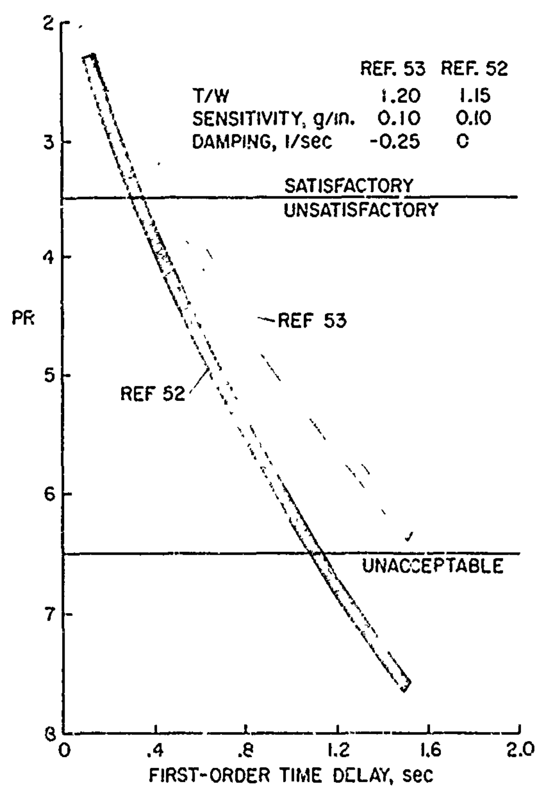


Figure 95. Variation of Pilot Rating With Thrust Response Time Delay for Vertical Touchdowns (Ref. 56).

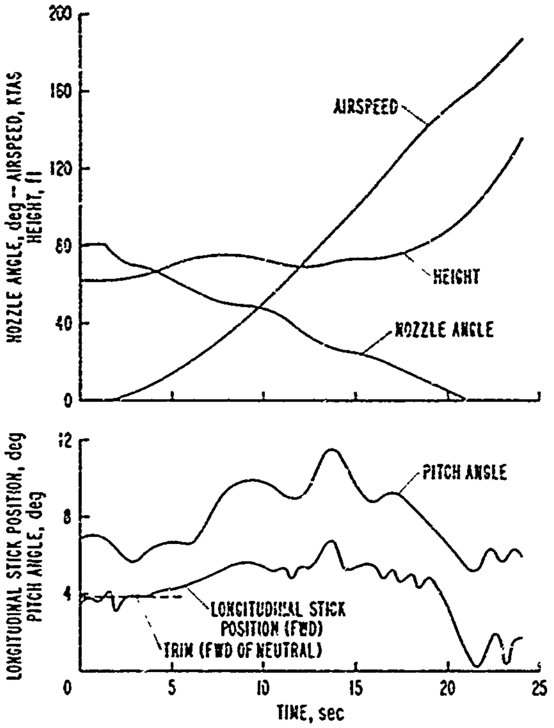


Figure 96. Time History of an Accelerating Transition, P1127 Aircraft (Ref. 7).

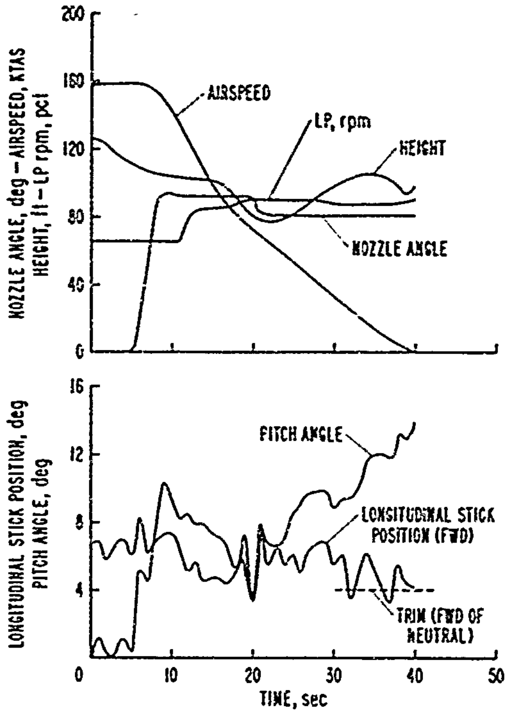


Figure 97. Time History of a Decelerating Transition, P1127 Aircraft (Ref. 7).

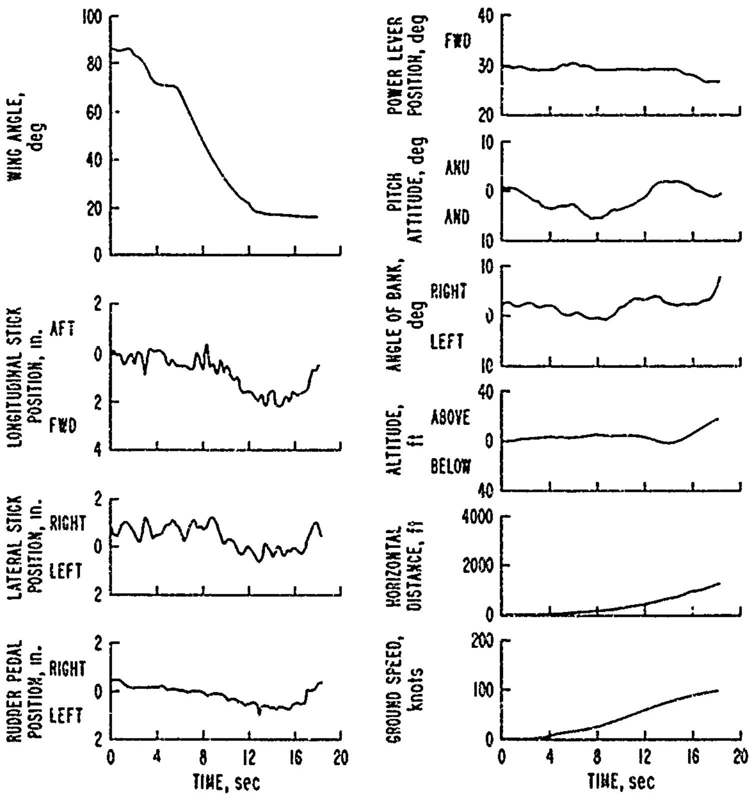


Figure 98. Time History of an Accelerating Transition, CL-84 Aircraft (Ref. 6).

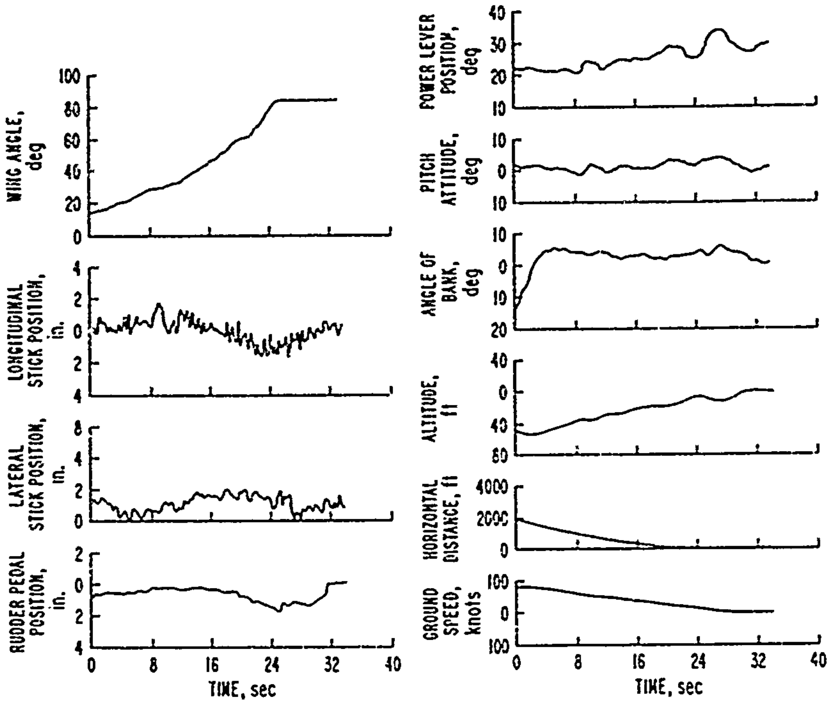


Figure 99. Time History of a Decelerating
Transition, CL-84 Aircraft (Ref. 6).

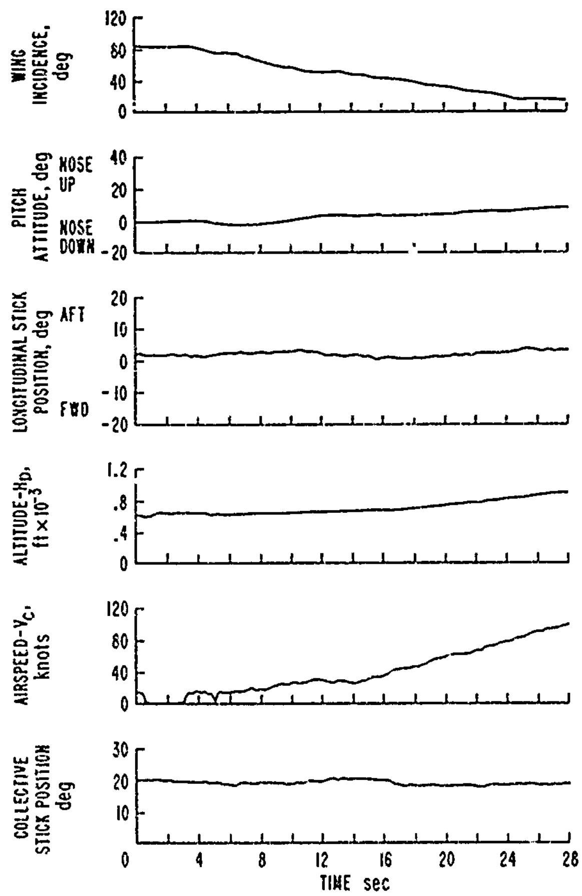


Figure 100. Time History of an Accelerating
Transition, XV-5A Aircraft (Ref. 9).

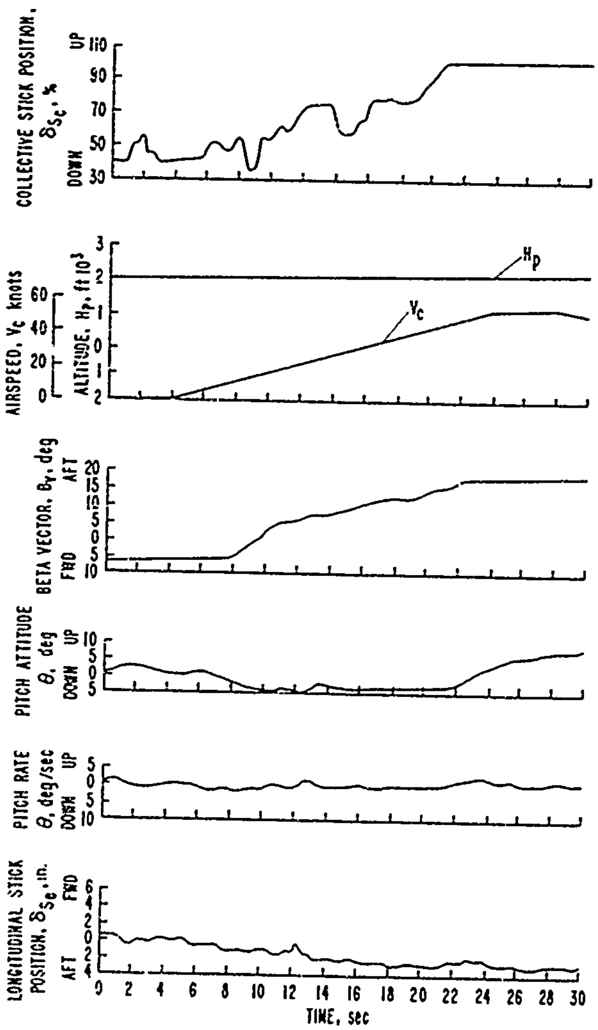


Figure 101. Time History of an Accelerating Transition, XC-142 Aircraft (Ref. 5).